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7.5K lbf Thrust Engine Preliminary Design for Orbit Transfer Vehicle

Task D.5 Final Report

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7.5K lbf THRUST ENGINE PRELIMINARY DESIGN FOR ORBIT TRANSFER VEHICLE

FINAL REPORT

TASK D.5

CONTRACT NAS 3-23772

Prepared For:

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NASA Lewis Research Center
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FOREWORD

This report documents the analysis, trade studies, and design work undertaken to define a 7500 lbf thrust LO_2/LH_2 rocket engine for use on an Orbital transfer Vehicle (OTV). Design work was completed to the preliminary level for the engine and components to the extent permitted by the task resources. Results of the study are expected to be very useful in any subsequent engine scaleup from the 7.5K lbf level. The report presents the analytical validation of the dual expander cycle engine as a high performance alternative to the hydrogen expander cycle engine, something not done in earlier work on this program.

The Engine Preliminary Design, Task Order C.5 to Contract NAS 3-23772, was completed in May 1988 with a design review at NASA Lewis Research Center. There is no intent to complete the design at the 7.5K lbf thrust level. Also included is a summary of the work done under Task Order C.3, Thrust Chamber Final Design. This task was halted just prior to completion of the preliminary design when there was a concern that additional design work at the 7.5K lbf thrust level would not benefit the expected Pathfinder Program mission needs. Sufficient work was completed to provide thrust chamber design data needed in the engine preliminary design task.

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LIST OF FIGURES (cont.)

1.0 SUMMARY

The objective of the 7.5K lbf thrust engine preliminary design task was to complete the necessary design and analysis and document it by means of reports, drawings, and formal design reviews. This has been accomplished with the issuance of this report. Aerojet has proposed a unique variant on the hydrogen expander cycle in which both hydrogen and oxygen are heated to run their mechanically separate turbopumps (TPAs). This dual expander cycle allows chamber pressure to be increased to 2000 psia which aids in meeting the 10:1 throttling requirement. An important bonus is the elimination of a helium purge system for the engine. The power balance work shows the design capable of operating over the mixture ratio and thrust range with an overthrust capability at high mixture ratios. Increased chamber material's temperatures below 500 psia and above 2000 psia chamber pressure would require limiting operation at these points to extend engine life. The proposed controls have sufficient range to handle all mixture ratio and thrust requirements, but the tank head start sequence is questionable until demonstrated with actual engine components due to the small (15 psi) pressure budget available. Component design was uncomplicated for valves but required some innovation for both turbopump and engine gimbal. The turbopumps use hydrostatic bearings to meet life requirements, and have a semi-spherical shape to handle pressure balance and thrust loading requirements. The oxygen TPA uses materials (primarily monel alloys) identified in an earlier NASA funded program for compatibility with 400°F oxygen. The gimbal requirement of ±20° in pitch and yaw seriously compromised component packaging and extendible nozzle design until it was divided into a ±6° throat gimbal and a 14° engine out gimbal. ATC also proposes designs for a hydrogen regenerator and LOX/GH_2 heat exchanger using their platelet structures technology for unusually small, lightweight components. The design for the extendible/retractable nozzle uses an electric motor driven jackscrew system with a single radiation cooled columbium or carbon-carbon nozzle section. The nominal engine design weight is 298 lbm (gimballed system). There are no pneumatic or purge gas requirements, and power consumption is 784 watts or less for normal operation. Preliminary reliability analyses and a risk assessment section complete the design task.

2.0 INTRODUCTION

2.1 BACKGROUND

This rocket engine design task supports the NASA-OAST plans for development of a new Orbit Transfer Vehicle (OTV) to be operational in the late 1990's. The current requirement for a space-based vehicle with payload capabilities greater than available upper stages results in a performance emphasis that mandates a new LO_2/LH_2 rocket engine representing a significant advance over presently available engines. Table 2-1 compares the characteristics of the RL-10 engine (the current U.S. manufactured engine) with those characteristics needed for the OTV engine. The RL-10 was a laudable achievement for the rocket engine propulsion technology of the 1960's, but the state-of-the-art had advanced enough by the 1980's for a new definition of requirements. The first attempt was presented in 1986 and is summarized in the second column. The most recent requirements are given in the third column. The OTV engine requirements pose a major challenge to the propulsion technology of the 1980's. In particular, the specific impulse, the engine life, and the throttling requirements are unprecedented and dictate advances in state-of-the-art.

Aerojet TechSystems Company (ATC) has worked with the NASA on technology contracts related to the OTV engine and its predecessors for over a decade. Several engine concepts were developed during this period that were rendered obsolete by changing requirements and technology. At no time was an ATC engine concept developed to the status of a preliminary design prior to the NASA LeRC establishing a new thrust level of 7500 lbf per engine under Contract NAS 3-23772. This also coincided with the completion of the Phase "A" prime contractor studies for the vehicle which aided in defining the engine envelope and operating requirements. This apparent increased definition and stability in the design requirements was considered justification for the engine preliminary design task reported herein. The task was started in May 1987 and completed in May 1988.

2.2 AEROJET DUAL EXPANDER ENGINE CYCLE

In a conventional expander cycle engine, hydrogen is routed through passages in the combustion chamber wall where it both cools the wall and acquires sufficient thermal energy to power the turbine drives of pumps for both the hydrogen and oxygen flow circuits. It is then routed to the injector for combustion. This cycle is fairly simple, plumbing is straightforward, and it offers good performance potential. Since all propellant is burned in the combustion chamber it does not have the losses of open cycles. Its limitations are related to dependence on only one

TABLE 2-I

TECHNOLOGY GOALS FOR THE NEW OTV ENGINE

Parameters	Reference Engine System Characteristics	October 1986 NASA Goals and/or Requirements	1988 Updated Goals or Requirements
Basing	Earth	Not Specified	Space
Human-rating	No	Not Specified	Yes
Design Criteria	Not Specified	Not Specified	Fail Operational, Fail Safe
Propellants - Fuel	Hydrogen	Hydrogen	Hydrogen
- Oxidizer	Oxygen	Oxygen	Oxygen
Vacuum Thrust (Design Point)	15,000 lbF	10,000 - 25,000 lbF*	7500 lbF (per engine)
Number of Engines per Vehicle		Not Specified	2 Minimum
Engine Mixture Ratio, O/F (Design Point)	5.0	6.0	6.0
Engine Mixture Ratio Range, O/F	4.4 to 5.6	5 - 7	5 - 7
Propellant Inlet Temperature -			
Hydrogen	38.3°R	37.8°R	37.8°
Oxygen	175.3°R	1762.7°R	162.7°R
Gimbal	±4.0 Degrees	±6.0 Degrees (Sq. Pattern)	±20 degrees Pitch & Yaw
Aerobraking Design Criteria	The engine must be compatible with aeroassist return of the vehicle to low-earth orbit.		
Vacuum Specific Impulse	444 lbF-sec/lbm	520 lbF-sec/lbm	490 lbF-sec/lbm
Vacuum Thrust Throttling Ratio	No Throttling	30:1	10:1
Net Positive Suction Head (NPSH) -			
Hydrogen	133.0 ft-lbf/lbm	0	15 ft-lbf/lbm
Oxygen	16.7 ft-lbf/lbm	0	2 ft-lbf/lbm
Weight	290 lbm	360 lbm	360 lbm (2 engines)
Length	70.1 in.	40	TBD (Assume 60", Stowed)
Reliability (90% Confidence Level)	0.9982	1.0	.9975 Single engine, .99958 Dual engine**
Operational Life	3 Starts, 4000 sec	500 Starts, 20 Hours	500 Starts, 20 Hours
Service Free-Life		100 Starts, 4 Hours	100 Starts, 4 Hours
Start Cycle		Chilldown with propulsive dumping of propellants, tankhead start, pumped idle operation, autogenous tank pressuration required	

Updated 4 January 1988

*Vehicle engine set total thrust must be in this range

**MSFC/Boeing Vehicle Study

working fluid for turbine drive energy. High turbopump and engine chamber pressures require the hydrogen to exit the regen chamber at temperatures very near to the design limits for the chamber liner. For copper alloy based chambers this is 1000°F. A marked improvement in engine operating flexibility and high chamber pressure capability is available if oxygen can be used as a working fluid driving the turbine on the oxidizer circuit TPA. This reduces the demands on the hydrogen circuit and allows for the turbopumps to be designed without interpropellant seals or mechanical geartrain connections.

The use of oxygen as a turbine drive fluid poses two problems; (1) where and how to add heat to the oxygen, and (2) selection of materials that will be compatible with the very reactive hot oxygen. The first ATC OTV engine design proposed an oxygen cooled centerbody inside an annular combustion chamber with a hydrogen cooled jacket in a 3000 lbf thrust engine. This was an aesthetically pleasing and apparently uncomplicated design concept. Scaleup to the 7500 lbf thrust level proved to be impractical, however, and a design change was required. The present engine concept has a two stage oxygen heating system. Cold oxygen liquid from the turbopump is initially vaporized and heated in a heat exchanger using waste heat from the hydrogen TPA turbine exit flow. Additional heat is added by flowing the oxygen through the regeneratively cooled nozzle extension. This 400°F oxygen has adequate energy to power the oxygen turbopump. The effectiveness of the dual expander cycle is evident in the high chamber pressure available (2000 psia) and the ability to operate the engine without a helium or other inert gas purge.

2.3 SCOPE OF WORK

The scope of work was to conduct a design and analysis program to provide a preliminary design of an expander cycle rocket engine concept which will be man-rateable, space maintainable and have variable thrust for use into a future reusable "Orbit Transfer Vehicle."

2.4 SPECIFIC SUBTASKS

The scope of work was to be accomplished by the performance of the following specific subtasks:

Subtask 1 - Work Plan

The procedure was to furnish to NASA LeRC, for approval and concurrence, a Program Work plan, including requirements, specifications, and master schedules.

Subtask 2 - Design

The preliminary design of the Aerojet OTV engine concept was accomplished within the guidelines, operational characteristics, and requirements listed in column 3 of Table 2-1. The OTV engine start cycle was as given in Figure 2-1. The preliminary design effort included the following:

1. Engine assembly layout drawing defining component arrangement, principle dimensions, and gimballed envelope.

2. Engine system layout drawing defining line routing and vehicle "proposed" interface requirements.

3. Engine assembly weight, center of gravity, and mass moment of inertia about the gimbal axes.

4. Component layout drawings showing principle dimensions and materials. Components are defined as major subassemblies such as the thrust chamber, pump with drive, or valves.

5. Component weights.

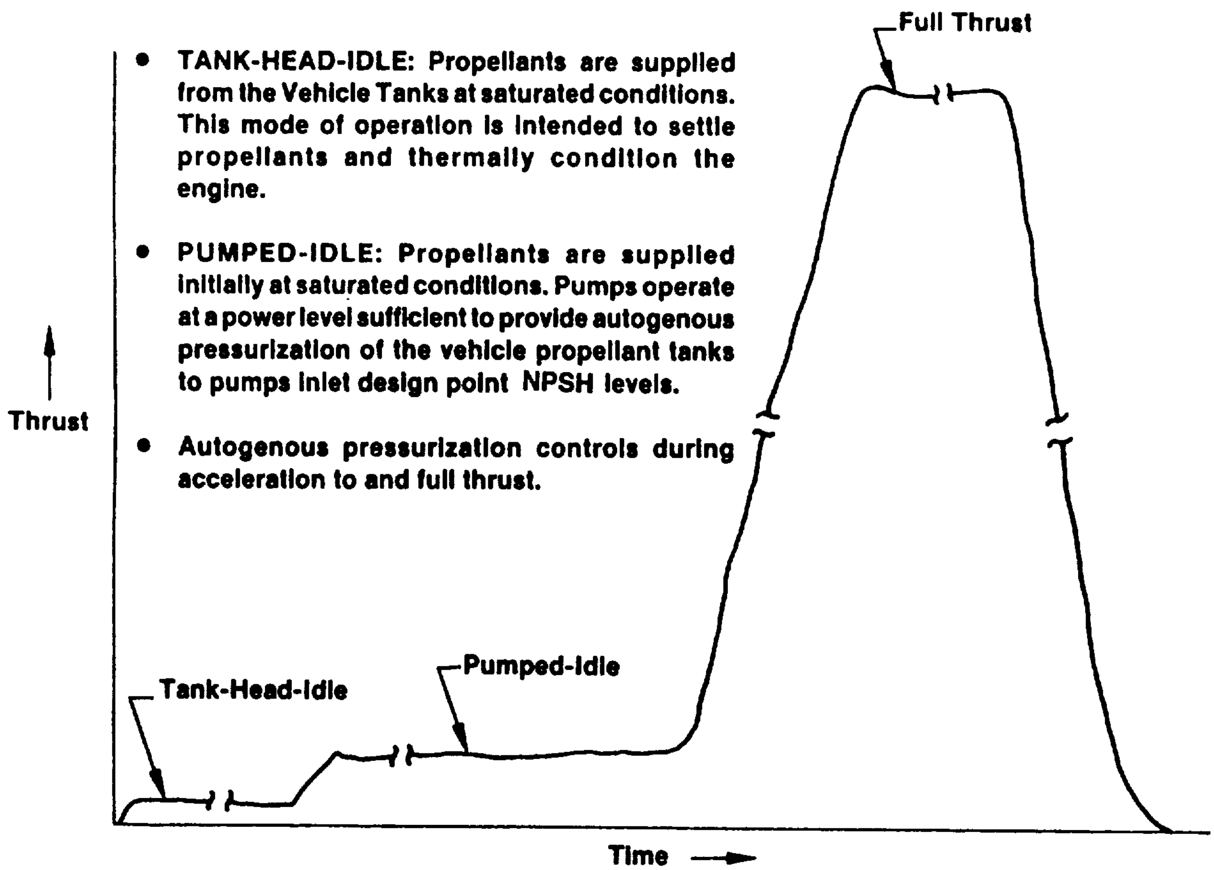
6. Engine cycle balance and performance analyses at design thrust over a mixture ratio range encompassing the design point mixture ratio plus and minus 1.0, including a detailed description of the analytical program used to predict engine system performance.

7. Description of engine operation and control, including a detailed description of the analytical program used to predict engine subsystem performance.

8. Electric power, purge, and pneumatic requirements.

9. Design and off-design performance analyses of the turbomachinery, including a detailed description of the analytical program used to predict pump drive performance.

10. Stress analyses, dynamic analyses, and fatigue analyses of critical parts, including a detailed description of the analytical program used to predict stress, study dynamics, and predict fatigue of the critical parts.



Engine Start to Full Thrust is to be Accomplished Using Tank-Head-Idle and Pumped-Idle Operating Modes as Shown Above

Figure 2.-1 OTV Engine Start Cycle

2.5 APPROACH AND GROUNDRULES

2.5.1 Preliminary Design Task Definition

A preliminary design is that design activity conducted prior to the detail design that defines the components, the general layout and a design approach. It includes sufficient analysis to predict both component and system performance. The design performance must be predicted to meet or exceed specified requirements. When performance goals are stated rather than requirements, the analysis should determine a numerical value to realistically assess the goal.

2.5.2 Work Package Allocation

Table 2-2 lists the various substasks by title and allocated hours. Design and analysis tasks were to be completed within this hour allocation whenever possible. A "best effort" approach was used so that a usable product would be generated without an overrun. In general this worked very well, but some subtasks had little depth as the hours did not allow for any extensive analysis or design iteration. Two subtasks were added that were funded from the program reserves: 1) the design of the LOX/GH_2 Heat Exchanger, and 2) the completion of a risk assessment section.

TABLE 2-2

WORK PACKAGE HOUR ALLOCATION

WORK PACKAGE HOUR ALLOCATION

SUB-TASK NUMBER	WORK PACKAGE TITLE	HOURS (INITIAL)	TASK PERCENT
I	Work Plan	60	1.5
IIa	Hydrogen TPA Design	360	9.0
IIb	Oxygen TPA Design	300	7.5
IIc	TCA Flt. Wt. MODS.	80	2.0
IId	Gimbal Design	80	2.0
IIe	Regenerator Design	160	4.0
IIf	Valve Design	320	8.0
IIg	Health Monitor Sensors	80	2.0
IIh	Component Integration	800	20.0
IIi	Electrical/Purge Requirements	80	2.0
IIj	Weight Center of Mass	80	2.0
IIk	Engine Power Balance	160	4.0
IIl	Nozzle Design	206	5.1
IIm	Nozzle Deployment System	114	2.9
IIn	Engine Operation & Control	120	3.0
IIIa	Reports & Management*	840	21.0
IIIb	Reviews	160	4.0
		4000	100.0

*Included All Program Reserves

3.0 DISCUSSION OF RESULTS

3.1 POWER BALANCE

A rocket engine power balance is the energy and mass balance computed at a particular design point (chamber pressure and mixture ratio) to verify that there is sufficient energy available to meet pressure requirements of the system without exceeding component temperature, pressure, and flow rate limitations. If the energy requirements of the system are greater than the available thermal energy there is no balance and the selected design point is unobtainable without changing the input conditions. Power balances for several design points are needed to map the engine performance envelope.

3.1.1 Power Balance Development

The detailed engine cycle schematic used to begin the power balance is given as Figure 3.1-1. Cycle restrictions used in the computer program are given in Table 3.1-1. The program development logic is given in Figure 3.1-2. The separate routines that were developed are named in Figure 3.1-3 and iteration loops are shown. Examples of the modeling techniques for individual components are presented in Figure 3.1-4. The operation of each component is simulated over the full range of operating conditions from chamber pressures of 200 to 2000 psia. The code's objective is to determine the operating parameters which result in a power balance for the engine while simultaneously constraining those parameters within their desired range of operation.

The temperature and pressure changes across passive components such as the baffles, heat exchanger, injector, or regeneratively cooled jacket are scaled from point design results of the baseline engine. The pumps and turbines use efficiency curves for off-design operation of the baseline component designs.

For a desired chamber pressure and mixture ratio, the system operating parameters are iterated until a closed solution is obtained. The independent parameters include turbine flowrates, pump discharge pressures, and turbine inlet temperatures.

Several subroutines commonly used at Aerojet for power balance programs involve representations of pump design curves. For this power balance the 7.5K preliminary TPA design curves were used for Q/N, efficiency, etc.

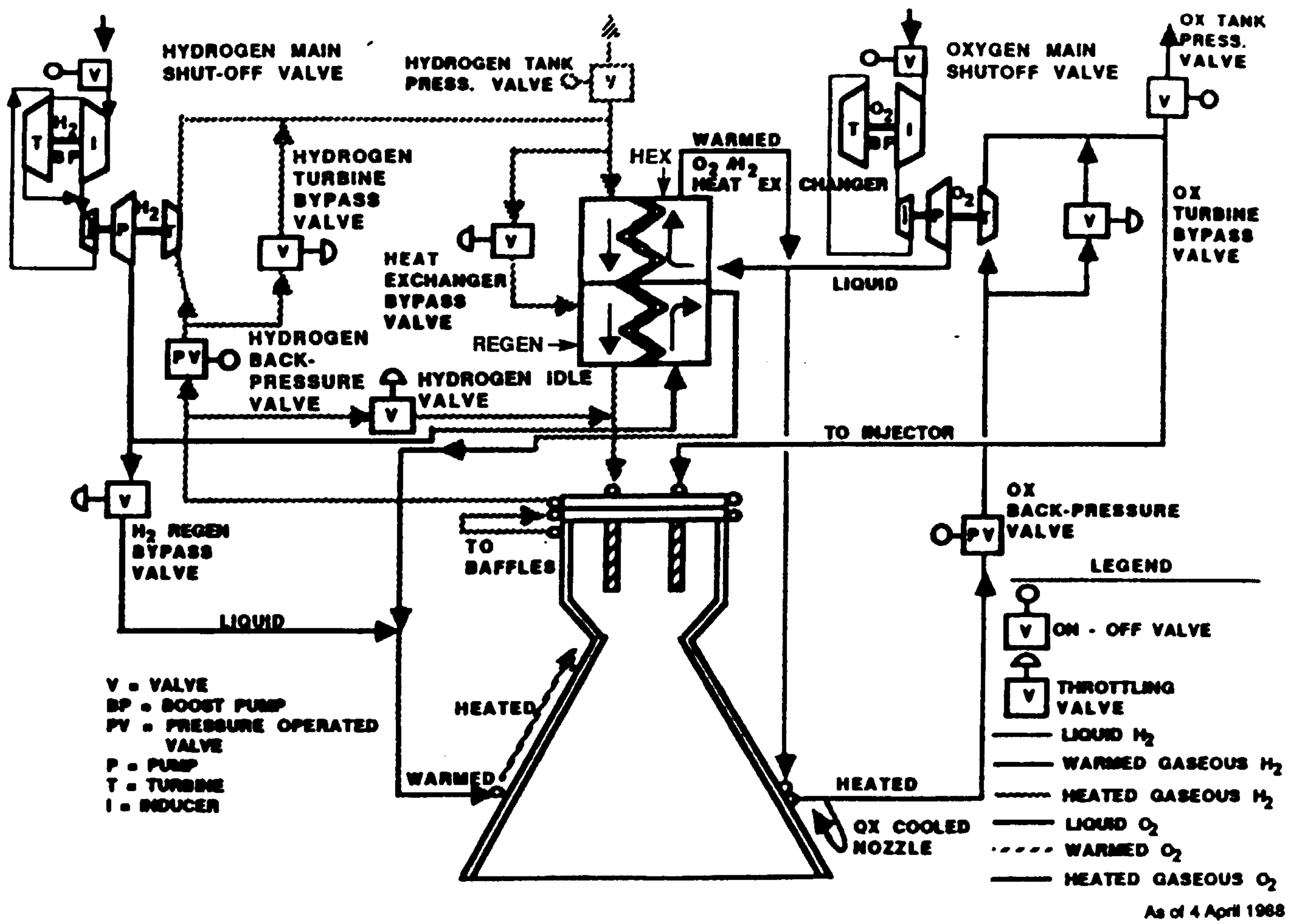


Figure 3.1-1 OTV Engine Dual Expander Cycle

TABLE 3.1-1

DUAL EXPANDER CYCLE POWER
BALANCE RESTRICTIONS

	Minimum		Maximum	Nominal	
Propellant Flow Rates					
Hydrogen Circuit	0	MR =	2.3 lbm/sec	2.188 lbm/sec	MR =
Oxygen Circuit	0	7.0	16.2 lbm/sec	13.785 lbm/sec	6.3
Propellant Temperatures					
Hydrogen	37.8°R		1710°R	N/A	
Oxygen	162.7°R		860°R	N/A	
Turbine Inlet Temperatures					
Hydrogen	N/A		1710°R	N/A	
Oxygen	N/A		860°R	N/A	
Wall Temperatures					
Thrust Chamber Throat			900°F	700°F	
Baffle Plate Wall			1000°F	800°F	
Turbine Bypass Percentage					
Hydrogen	0*		100	6**	
Oxygen	0*		100	10**	
Hydrogen Temperature from Hydrogen Regenerator	~100°R		400°R	400°R**	

*Overthrust Operation Only
**At Full Thrust, MR = 6

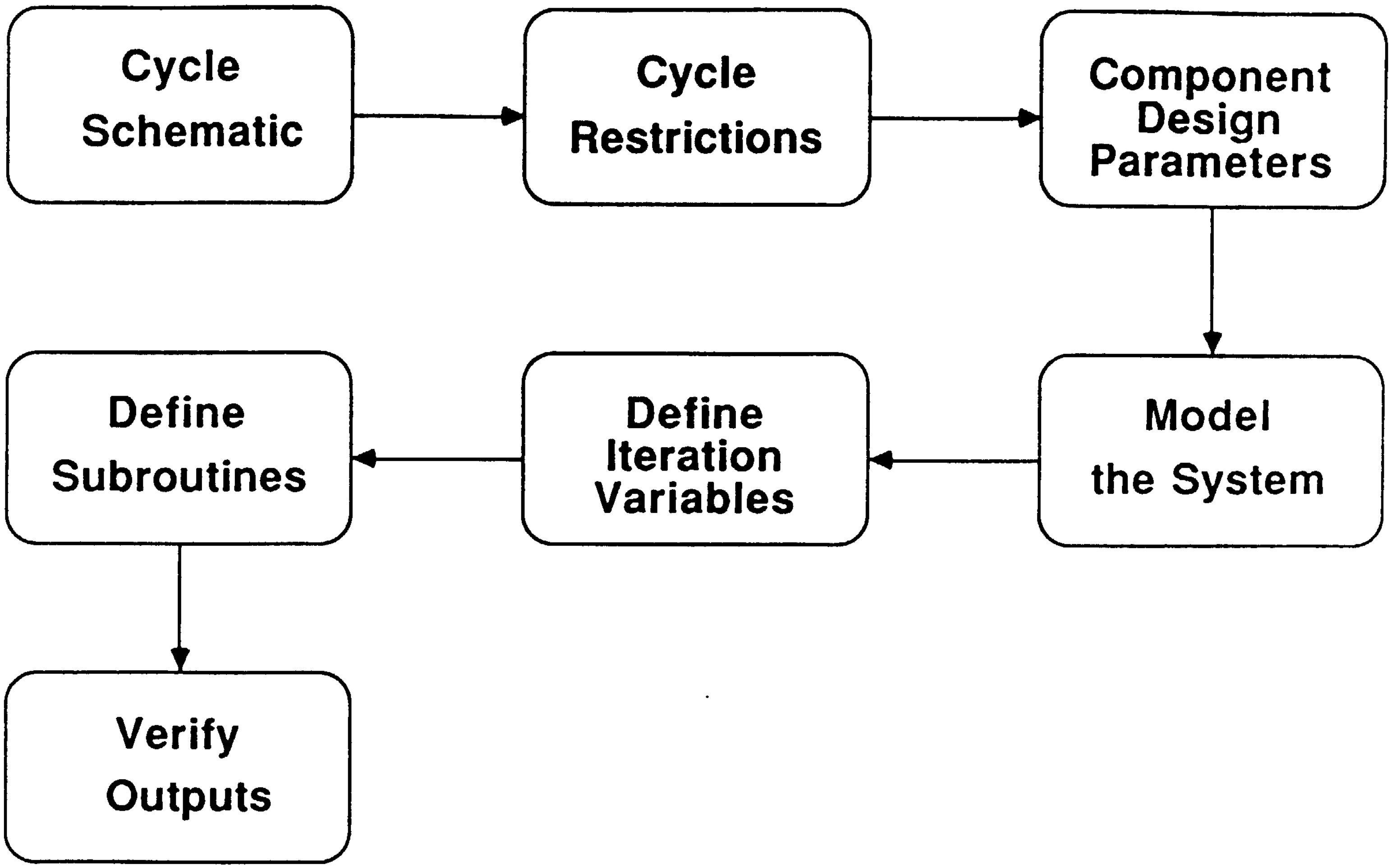


Figure 3.1-2 Computerizing a Power Balance

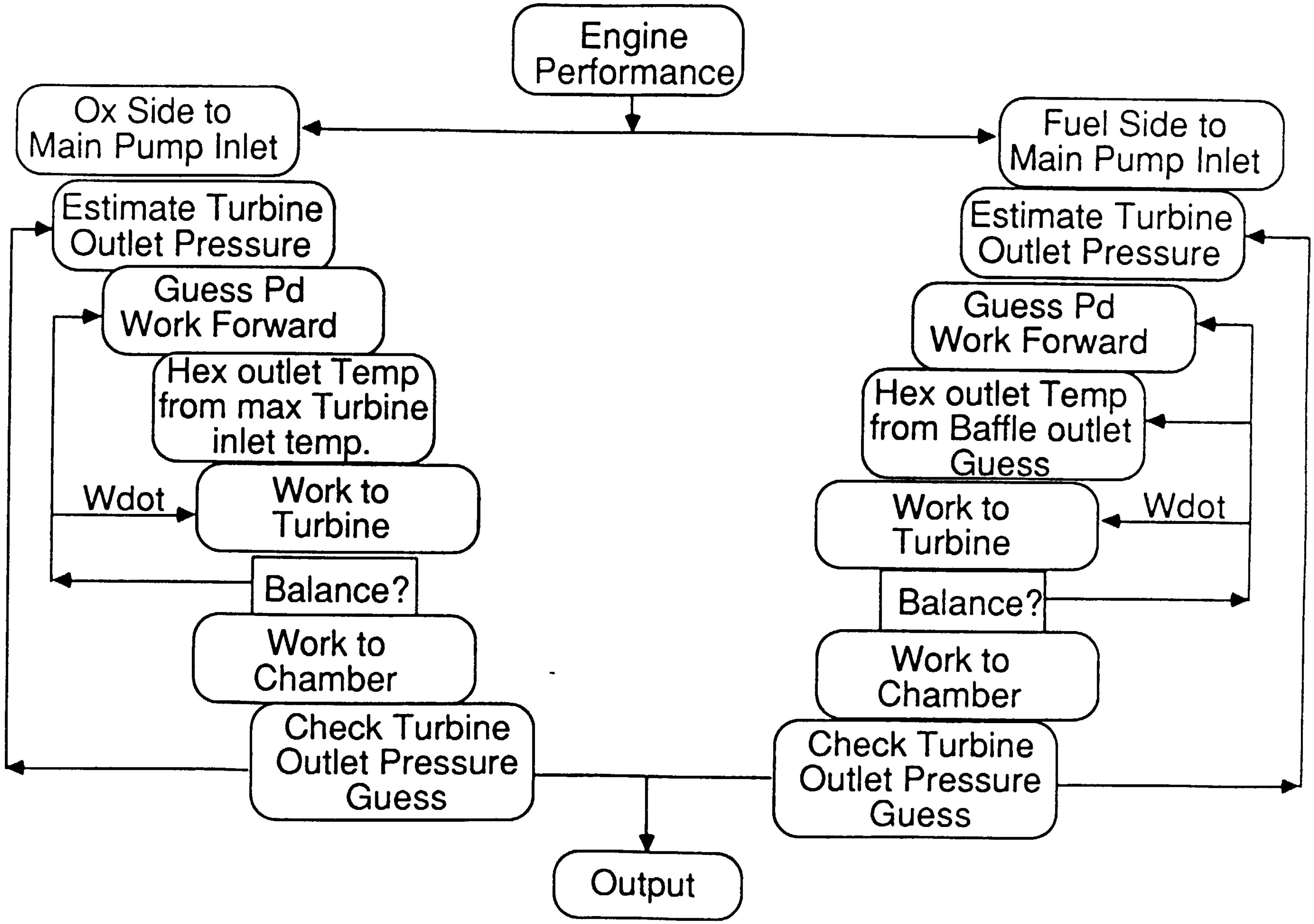


Figure 3.1-3 Power Balance Program Iterations

MAIN PUMP

Head = ΔH*778

rpm=Q/N, H^2//N Curves

eff = eff Curves

HP = w*ΔP/(Rho*eff)

TURBINE

U = Dia*N*CNV

Co = (2*g*ΔHideal)**.5

eff = f(U,Co) = ΔHideal/ΔHactual

HP=w*eff*ΔHideal

HEAT EXCHANGER

Cool Side	Hot Side
Set Outlet Temperature	
Qc = wΔH	Qh = -Qc
25% Bypass	

VALVES

Set Flow Area (CdA)

Gas or Liquid Orifice Subroutine to calculate Outlet Pressure

Figure 3.1-4 Component Modeling

The output of the power balance program for the 2000 psia chamber pressure, MR = 6 case (nominal engine operation) is given as Table 3.1-2. This is a dense amount of information very useful to a designer but requiring considerable interpretation. Figures 3.1-5 and 3.1-6 are representations of the power balancer at the same point with the circuit flow path indicated and values for pressure, temperature, and propellant density shown after each component. These figures also demonstrate the use of the power balance as a design tool as they show the effect on the power balance of increasing the maximum baffle outlet temperature to 1552.6°R while holding a 1.78 inch hydrogen turbine wheel diameter (Figure 3.1-5) versus increasing the turbine wheel diameter to 2.35 inches (Figure 3.1-6). The chamber pressure and mixture ratio are the same, but there is a significant difference in component design and engine life. This type of trade can be done very quickly with the power balance program with readily entered input data changes.

3.1.2 Engine Operating Envelope

The operating envelope of interest is the area shown in Figure 3.1-7 where mixture ratio is plotted against engine chamber pressure. Runs were made for 17 points as indicated on the plot. An "X" indicates a non-balance point which is outside the engine operating envelope. The small circles are balanced points. No mapping was done below MR = 5 or above MR = 7 although the envelope could be extended into these MR regions.

The nominal design point (Pc = 2000, MR = 6) is bounded by several of the system constraints. The H_2 turbine bypass is at a minimum allowable, the regen chamber inlet temperature is at a maximum, the H_2 turbine inlet temperature is at a maximum. At this point, very little lowering of the MR or raising of the Pc can be accomplished.

A full 10 to 1 throttling range is available at MR = 6. For operation at low chamber pressure (< 500 psia) an MR of 5 or less is recommended to provide better thermal margins. Engine overthrust operation is possible at mixture ratios of 6 and greater. A recommended structural design point would be for a maximum chamber pressure of 2500 psia and whatever internal pressures were necessary to produce this limiting power. This would be a 20% increase over rated thrust or 9000 lbf.

Table 3.1-2

DTV ENGINE POWER BALANCE — Page 1

	Oxidizer			Fuel		
	R	= 579.70	(in/deg R)	R	= 9202.00	(in/deg R)
Tank Conditions	Pout	= 15.00	(psia)	Pout	= 20.00	(psia)
	Tout	= 162.70	(deg R)	Tout	= 37.80	(deg R)
	Hout	= -57.17	(BTU/#)	Hout	= -117.35	(BTU/#)
	Rho out	= 71.17	(#/cuft)	Rho out	= 4.34	(#/cuft)
Shut-off Valve	Pin	= 15.00	(psia)	Pin	= 20.00	(psia)
	Pout	= 15.00	(psia)	Pout	= 19.82	(psia)
	Delta P	= 0.66	(psi)	Delta P	= 0.18	(psi)
	T	= 162.70	(deg R)	T	= 37.80	(deg R)
	H	= -57.17	(BTU/#)	H	= -117.35	(BTU/#)
	CdA	= 3.01000	(in^2)	CdA	= 3.75000	(in^2)
Boost Pump Conditions	Pin	= 15.00	(psia)	Pin	= 19.82	(psia)
	Pout	= 37.90	(psia)	Pout	= 41.12	(psia)
	Tin	= 162.70	(deg R)	Tin	= 37.80	(deg R)
	Tout	= 162.70	(deg R)	Tout	= 38.07	(deg R)
	Rho in	= 71.1696	(lb/ft3)	Rho in	= 4.3356	(lb/ft3)
	Wdot	= 13.785	(lb/sec)	Wdot	= 2.188	(lb/sec)
	Eff (hyd)	= 0.700		Eff (hyd)	= 0.700	
	N	= 52832.53	(rpm)	N	= 122724.98	(rpm)
	HP	= 1.66	(HP)	HP	= 4.02	(HP)
	Q/N	= .0159445	(gpm/rpm)	Q/N	= .0113863	(gpm/rpm)
Pump Conditions	Pin	= 37.90	(psia)	Pin	= 41.12	(psia)
	Pout	= 5313.28	(psia)	Pout	= 4250.00	(psia)
	Tin	= 162.70	(deg R)	Tin	= 38.07	(deg R)
	Tout	= 188.21	(deg R)	Tout	= 116.55	(deg R)
	Rho in	= 71.1917	(lb/ft3)	Rho in	= 4.3382	(lb/ft3)
	Wdot	= 13.785	(lb/sec)	Wdot	= 2.188	(lb/sec)
	Eff (hyd)	= 0.740		Eff (hyd)	= 0.570	
	Eff (mch)	= 1.000		Eff (mch)	= 1.000	
	N	= 52832.53	(rpm)	N	= 122724.98	(rpm)
	HP	= 361.29	(HP)	HP	= 975.03	(HP)
	Q/N	= .0016449	(gpm/rpm)	Q/N	= .0018445	(gpm/rpm)
Ox: Cool Side Heat Exchanger	Pin	= 5313.28	(psia)	Pin	= 4250.00	(psia)
	Pout	= 5118.28	(psia)	Pout	= 4200.00	(psia)
	Delta P	= 195.00	(psi)	Delta P	= 50.00	(psi)
	Tin	= 188.21	(deg R)	Tin	= 116.55	(deg R)
Fuel: Cool Side Regenerator	Tout	= 580.90	(deg R)	Tout	= 746.19	(deg R)
	Delta T	= 392.69	(deg R)	Delta T	= 629.64	(deg R)
	Hin	= -38.59	(BTU/#)	Hin	= 198.99	(BTU/#)
	Hout	= 103.59	(BTU/#)	Hout	= 2593.19	(BTU/#)
	Qdot	= 1959.91	(BTU/s)	Qdot	= 3928.94	(BTU/s)
	Wdot	= 13.78	(#/sec)	Wdot	= 1.64	(#/sec)
				% bypass	= 25.00	

Table 3.1-2 (Cont)

OTV ENGINE POWER BALANCE — Page 2

	Oxidizer			Fuel		
Regen Jacket				Pin	= 4200.00	(psia)
				Pout	= 3626.00	(psia)
				Delta P	= 574.00	(psi)
				Tin	= 578.71	(deg R)
				Tout	= 937.51	(deg R)
				Delta T	= 358.80	(deg R)
				Hin	= 1994.63	(BTU/#)
				Hout	= 3266.71	(BTU/#)
				Wdot	= 2.19	(#/sec)
Baffles				Pin	= 3626.00	(psia)
				Pout	= 3550.80	(psia)
				Delta P	= 75.20	(psi)
				Tin	= 937.51	(deg R)
				Tout	= 1372.61	(deg R)
				Delta T	= 435.10	(deg R)
				Hin	= 3266.71	(BTU/#)
				Hout	= 4794.02	(BTU/#)
				Wdot	= 2.19	(#/sec)
Ox Nozzle Cooling	Pin	= 5118.28	(psia)			
	Pout	= 4758.98	(psia)			
	Delta P	= 359.30	(psi)			
	Tin	= 580.90	(deg R)			
	Tout	= 860.00	(deg R)			
	Delta T	= 279.10	(deg R)			
	Hin	= 103.59	(BTU/#)			
	Hout	= 181.13	(BTU/#)			
	Wdot	= 13.78	(#/sec)			
Back Pressure Valve	Pin	= 4758.98	(psia)	Pin	= 3550.80	(psia)
	Pout	= 4727.62	(psia)	Pout	= 3423.83	(psia)
	Delta P	= 31.36	(psi)	Delta P	= 126.97	(psi)
	T	= 860.00	(deg R)	T	= 1372.61	(deg R)
	H	= 181.13	(BTU/#)	H	= 4794.02	(BTU/#)
	CdA	= 0.91000	(in^2)	CdA	= 0.42500	(in^2)
Turbine Conditions	Pin	= 4727.62	(psia)	Pin	= 3423.83	(psia)
	Pout	= 2312.28	(psia)	Pout	= 2187.04	(psia)
	Wdot	= 12.406	(lb/sec)	Wdot	= 1.908	(lb/sec)
	Tin	= 860.00	(deg R)	Tin	= 1373.55	(deg R)
	Tout	= 765.59	(deg R)	Tout	= 1280.06	(deg R)
	Hin	= 181.16	(BTU/#)	Hin	= 4794.02	(BTU/#)
	Hout	= 160.67	(BTU/#)	Hout	= 4433.89	(BTU/#)
	Eff.	= 0.561		Eff.	= 0.595	
	HP	= 359.50	(HP)	HP	= 972.13	(HP)
	PR	= 2.045	(Pin/Pout)	PR	= 1.566	(Pin/Pout)
	U/Co	= 0.310	(fps/fps)	U/Co	= 0.229	(fps/fps)
	Wheel dia	= 1.820	(in)	Wheel dia	= 2.350	(in)
	%bypass	= 10.00		% bypass	= 12.78	

Table 3.1-2 (Cont)

OTV ENGINE POWER BALANCE									Page 3
	Oxidizer				Fuel				
Hot Side Heat Exchanger					Pin	=	2187.04	(psia)	
					Pout	=	2117.04	(psia)	
					Delta P	=	70.00	(psi)	
					Tin	=	1118.40	(deg R)	
					Tout	=	777.94	(deg R)	
					Delta T	=	340.46	(deg R)	
					Hin	=	3867.15	(BTU/#)	
					Hout	=	2672.82	(BTU/#)	
					Qdot	=	-1959.91	(BTU/s)	
					Wdot	=	1.64	(#/sec)	
					% bypass	=	25.00		
Gas Side Regenerator					Pin	=	2117.04	(psia)	
					Pout	=	2097.04	(psia)	
					Delta P	=	20.00	(psi)	
					Tin	=	862.87	(deg R)	
					Tout	=	367.69	(deg R)	
					Delta T	=	495.18	(deg R)	
					Hin	=	2971.00	(BTU/#)	
					Hout	=	1175.34	(BTU/#)	
					Qdot	=	-3928.94	(BTU/s)	
					Wdot	=	2.19	(#/sec)	
Injector	Pin	=	2312.28	(psia)	Pin	=	2097.04	(psia)	
	Pout	=	1998.26	(psia)	Pout	=	2001.65	(psia)	
	Tin	=	773.75	(deg R)	Tin	=	367.69	(deg R)	
	Wdot	=	13.128	(lb/sec)	Wdot	=	2.188	(lb/sec)	
	Drop	=	314.03	(psia)	Drop	=	95.40	(psia)	
	CdA	=	0.40000	(in^2)	CdA	=	0.33200	(in^2)	
Combustion Chamber	PC	=	2000.00	(psia)	MR	=	6.00	(O/F)	
	DPcc	=	1.49	(psia)	Wdot	=	15.32	(lb/sec)	
	ERE	=	1.000		Dthroat	=	1.518	(in)	
	F	=	7351.76	(lbf)	Isp	=	480.00	(sec)	

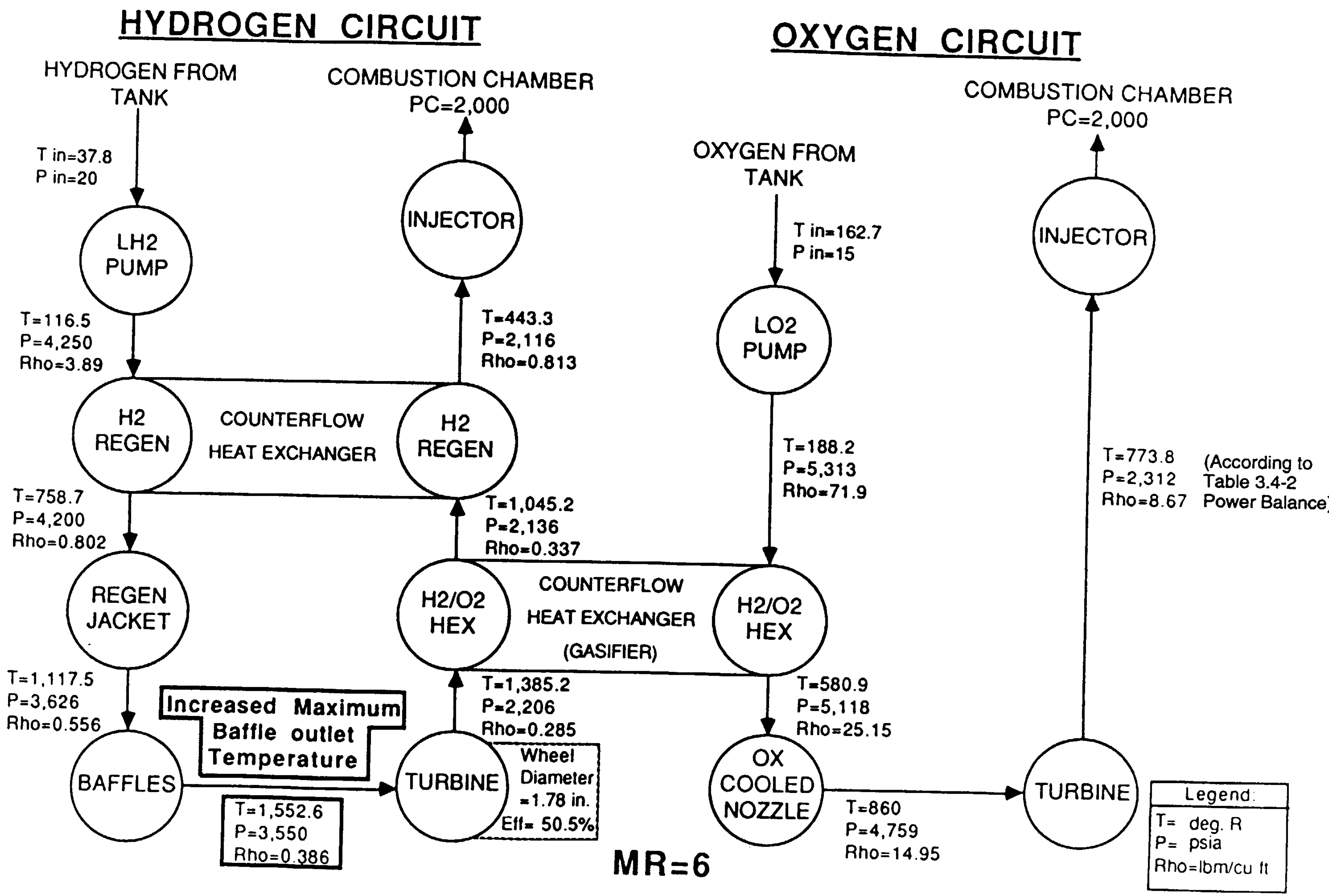


Figure 3.1-5 OTV Engine Preliminary Design Power Balance

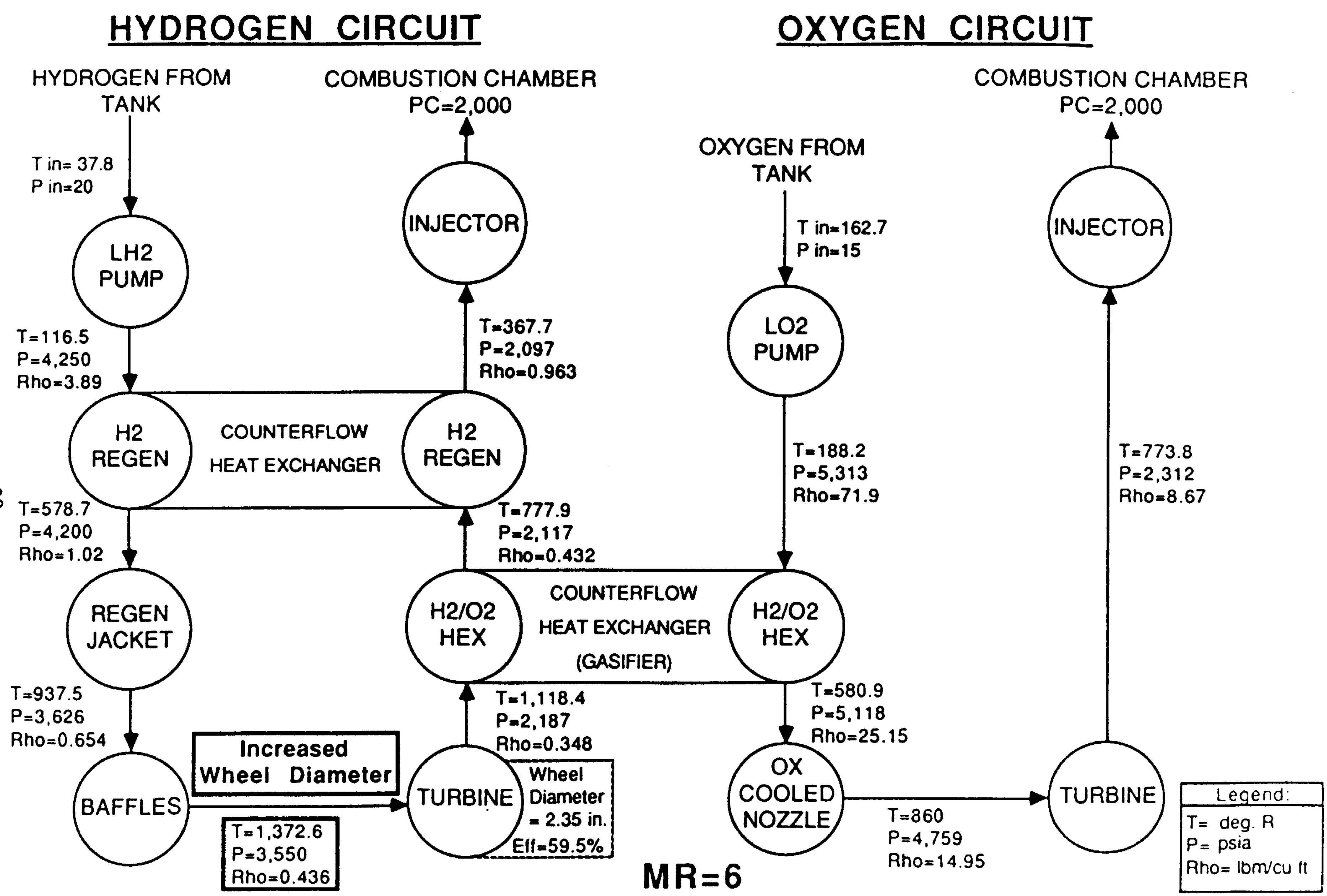


Figure 3.1-6 OTV Engine Preliminary Design Power Balance

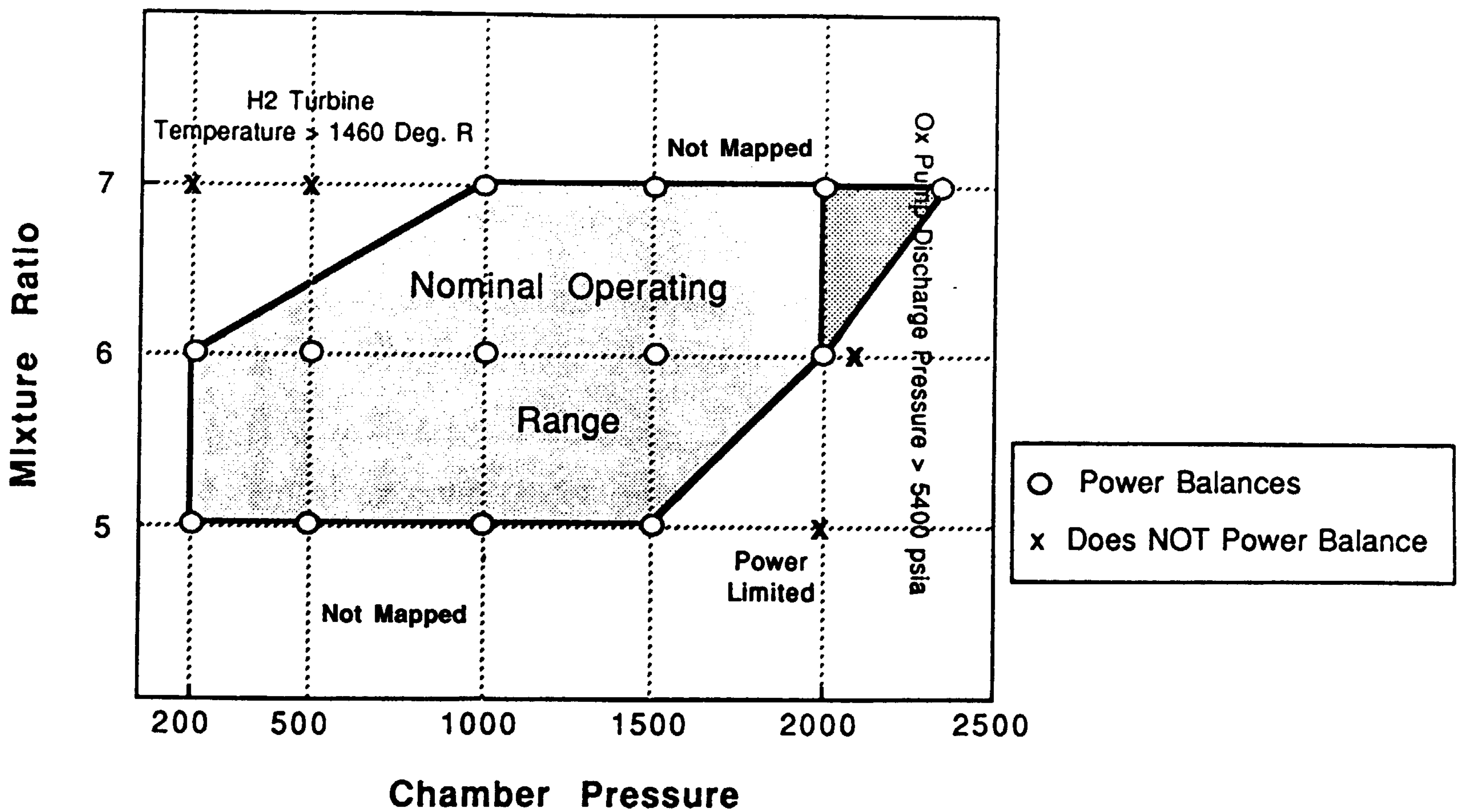


Figure 3.1-7 OTV Preliminary Design Engine Operating Envelope

3.1.3 Thermal Limitations

The high mixture ratio/low chamber pressure cases are limited by the turbine inlet temperature of the hydrogen gas. This, in turn, is a limit of the baffle plate copper wall on the hot gas side. At 1000°F the maximum hydrogen temperature is going to be about 900°F. This is indicated on Figure 3.1-8 which is a plot of entropy versus enthalpy for hydrogen gas. Note that enthalpy is very little affected by the system pressure. To get more enthalpy in the hydrogen the temperature must be raised; higher pump outlet pressures won't help as is shown in Figure 3.1-9. The copper baffle plate, then, is a limit to engine power. In an attempt to circumvent this limit a baffle plate concept was prepared that would allow a much higher hydrogen temperature to the fuel turbopump assembly (TPA) turbine. This concept is shown in Figure 3.1-9. It is based on a platinum section in the plate that can operate at over 2000°F wall temperature. The thermal limit now shifts to the hydrogen TPA hot section parts. For the present TPA design this is 1200°F. This 300 degree increase would significantly expand the operating envelope and engine flexibility. It was concluded by the design team that this would be a very valuable design change.

The low mixture ratio/high chamber pressure cases are limited by oxygen turbopump power availability. With an upper temperature limit set at 400°F for oxygen to the Ox TPA turbine this is another limit set by materials. The 400°F is a consensus figure derived from the oxygen-materials compatibility program for safe operation of the materials used in the TPA with oxygen. Until there is more experimental work with oxygen compatible materials the limit is unlikely to be raised.

There are also thermal limits at both the greater than 2000 psia chamber pressure region and the less than 500 psia chamber pressure region. The lower chamber pressures are accompanied by a bulk temperature rise in the hydrogen due to longer transit time through the regen chamber and baffles. The high chamber pressures are actually attained by adding waste heat from the hydrogen turbine exit stream to the hydrogen entering the regen chamber. This is done by the regenerator. It is limited, however, as the gas side wall temperature in the throat increases as the inlet hydrogen temperature increases. When the inlet hydrogen is at 400°R the wall is at 1260°R, the thermal limit. This was the design point for 2000 psia chamber pressure at MR = 6. The variation of hydrogen inlet temperature with thrust is given in Figure 3.1-10. The odd shape of the curve is partly due to the interaction with the hydrogen turbine bypass valve.

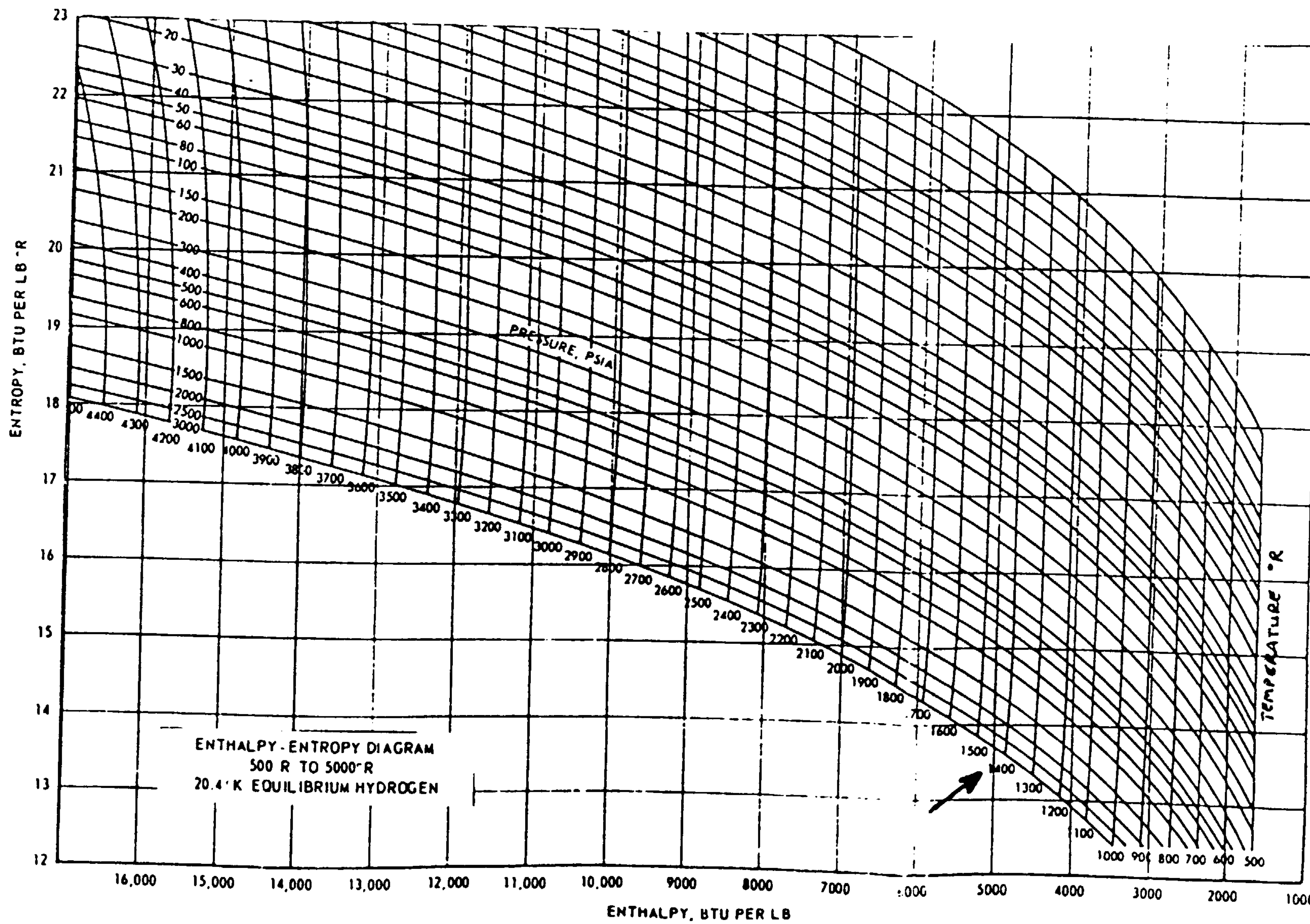


Figure 3.1-8 Enthalpy is a Function of Temperature

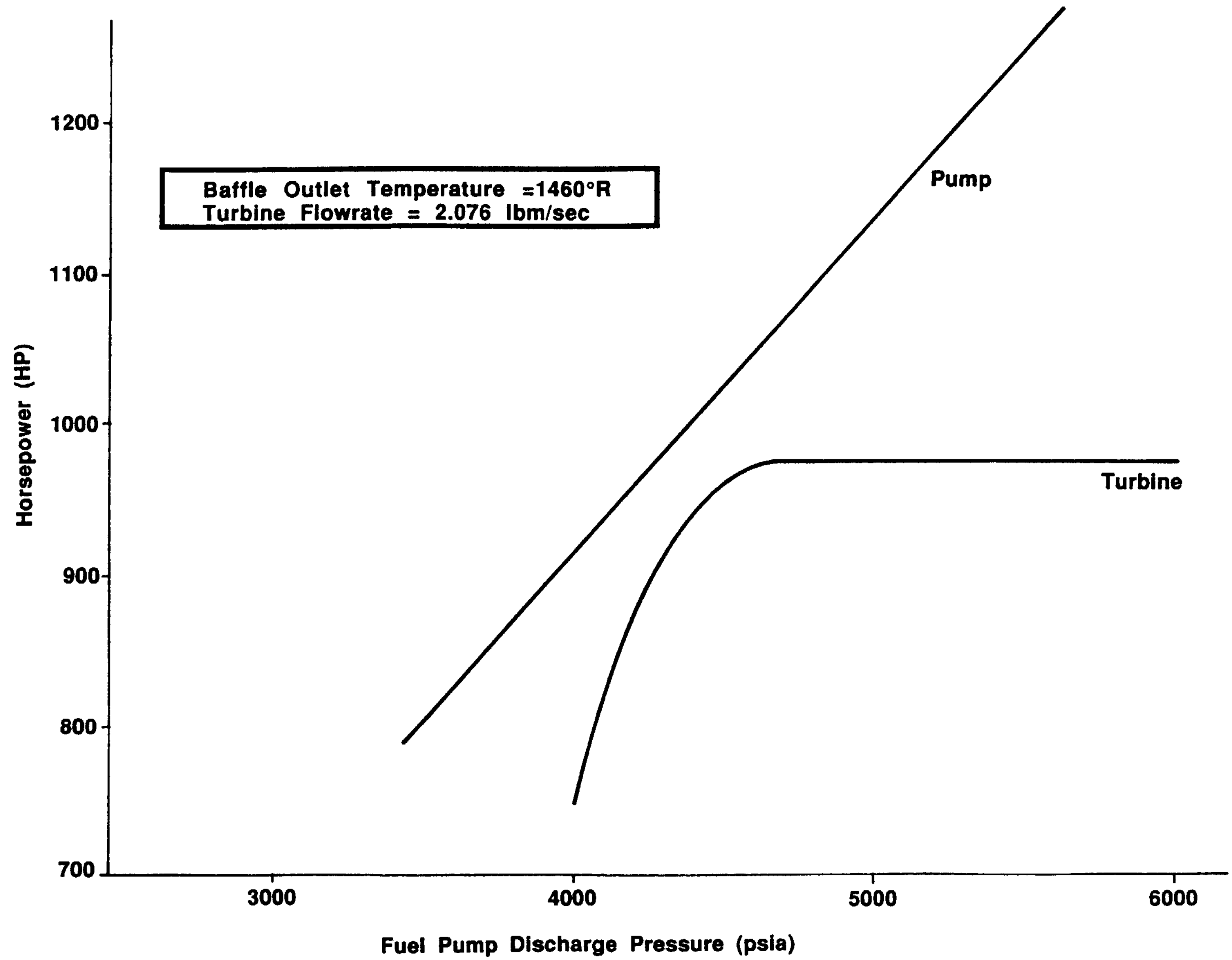


Figure 3.1-9 Fuel Pump Discharge Pressures Above 4,600 psia DO NOT Increase Fuel Turbine Horsepower

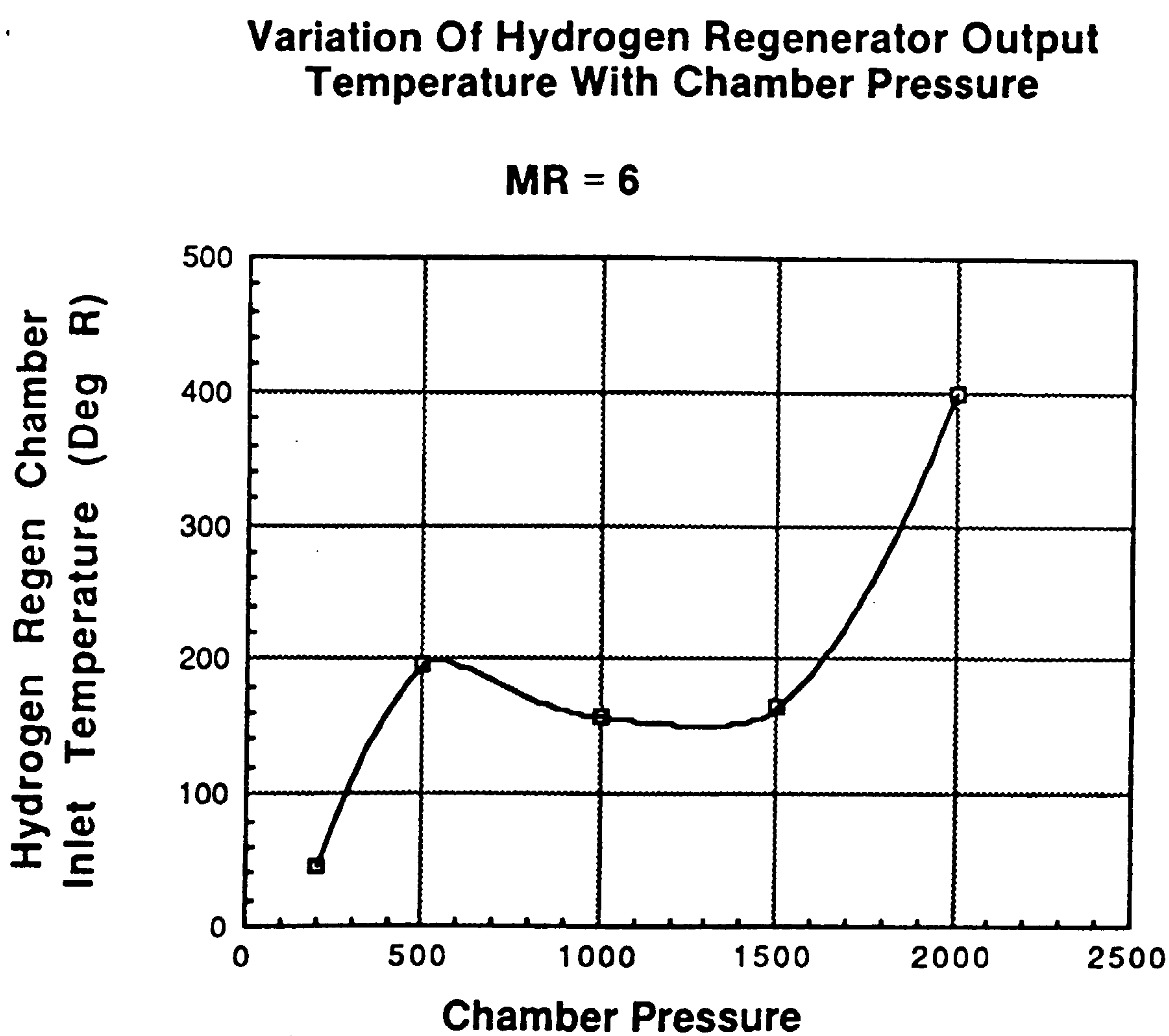


Figure 3.1-10 OTV Engine Preliminary Design Engine Operation

3.1.4 Control Effectiveness

The power balance provided information on the likely effectiveness of the primary controls for adjusting engine power and mixture ratio: the turbine bypass valves. Their operating condition over the full throttle range is given in Figure 3.1-11. The oxygen bypass valve is very nearly linear over the range and is ideal for control purposes. The hydrogen bypass valve has a very narrow operating range and a humpbacked curve. This shows the interaction with the hydrogen regenerator. Both valves must be programmed together for stable operation of the hydrogen circuit. Also, the hydrogen turbine bypass valve must be designed for vernier operation in the zero to 20% bypass range due to the sensitivity in this region.

One interesting feature of the engine is that there is a minimum temperature/maximum life operating point at a chamber pressure of 1500 psia. This corresponds to a thrust of 5625 lbf. An approach to long life would be to design the nominal thrust point at the minimum temperature point. This would give a 60% overthrust capability for the engine, but the overall engine would be somewhat larger and heavier than the present design. This minimum temperature design point is clearly seen in Figure 3.1-12 as the minima of the curve.

3.1.5 Performance Prediction

With early power balance work showing that the engine would balance at 2000 psia and MR = 6, performance calculations were made based on energy release efficiencies (ERE) of 99.5% and 100%. Figure 3.1-13 is a summary chart for the specific impulse (Isp) predictions for this design. The two curves are for chamber pressures of 2000 psia and 3000 psia at an ERE of 99.5%. The diamond was the actual performance prediction of this engine design at 99.5% ERE and an area ratio nozzle of 1560 to 1. The square was the performance for 100% ERE but for the same conditions otherwise.

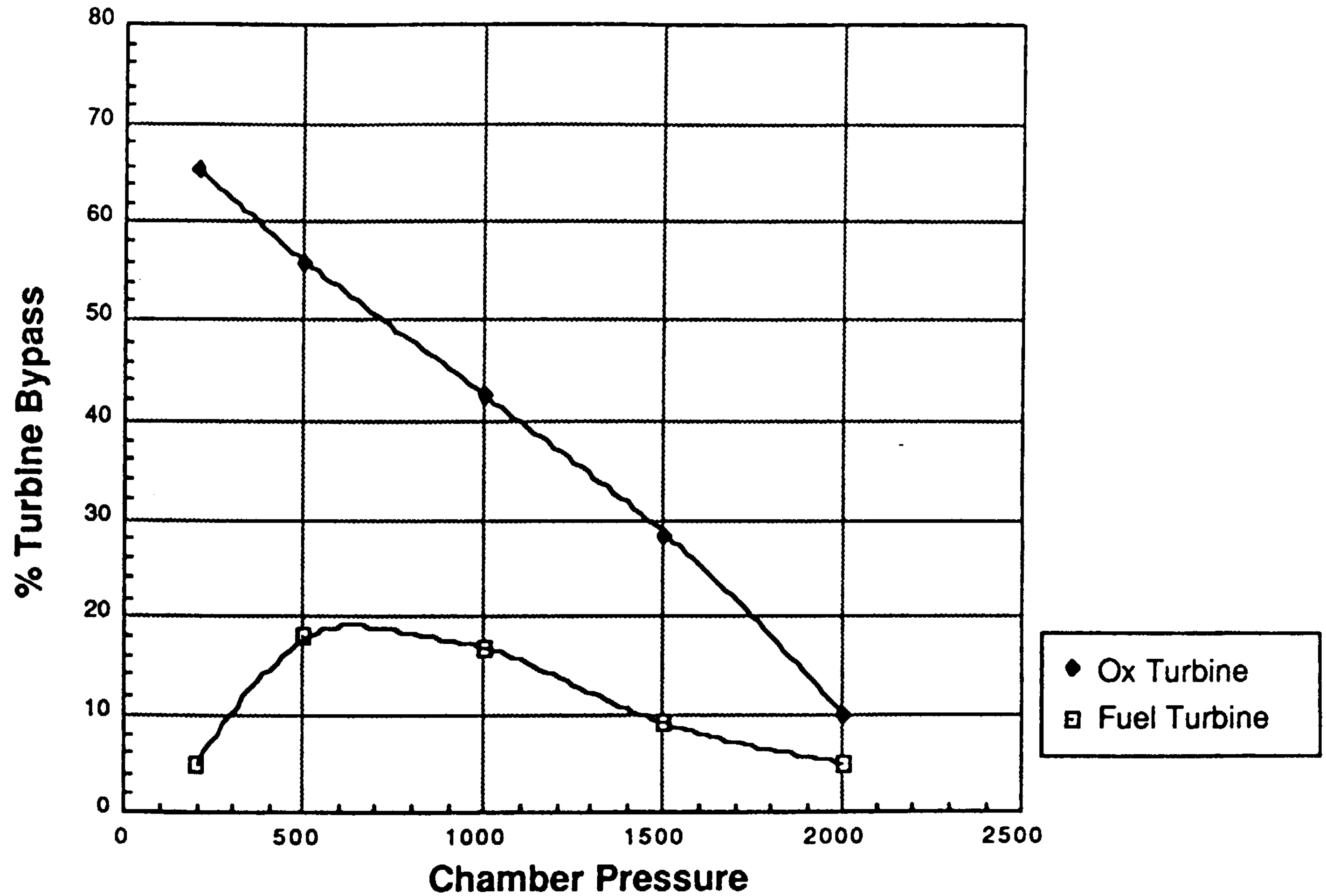


Figure 3.1-11 Turbine Bypass for OTV Engine (At MR=6)

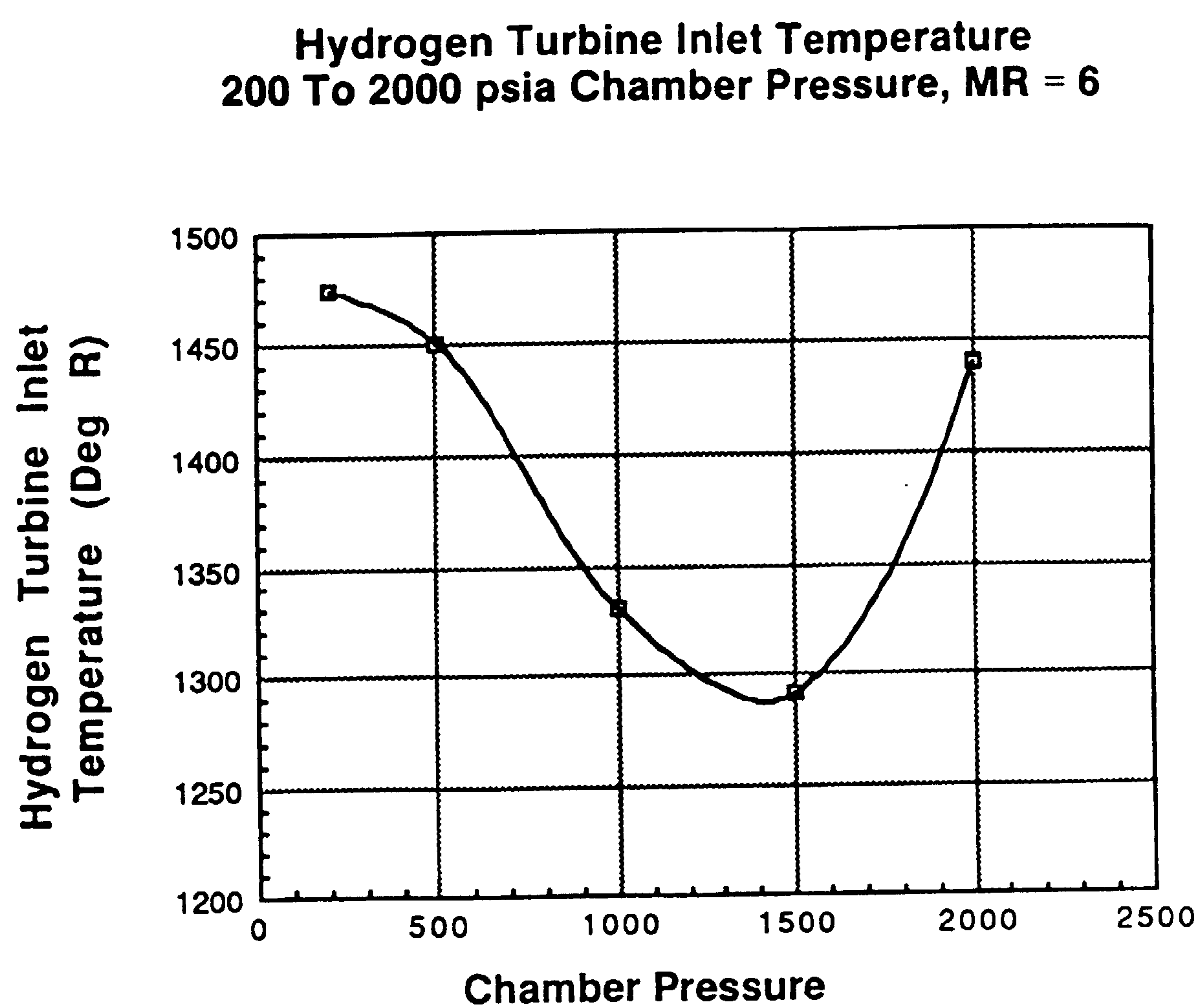


Figure 3.1-12 OTV Engine Preliminary Design Engine Operation

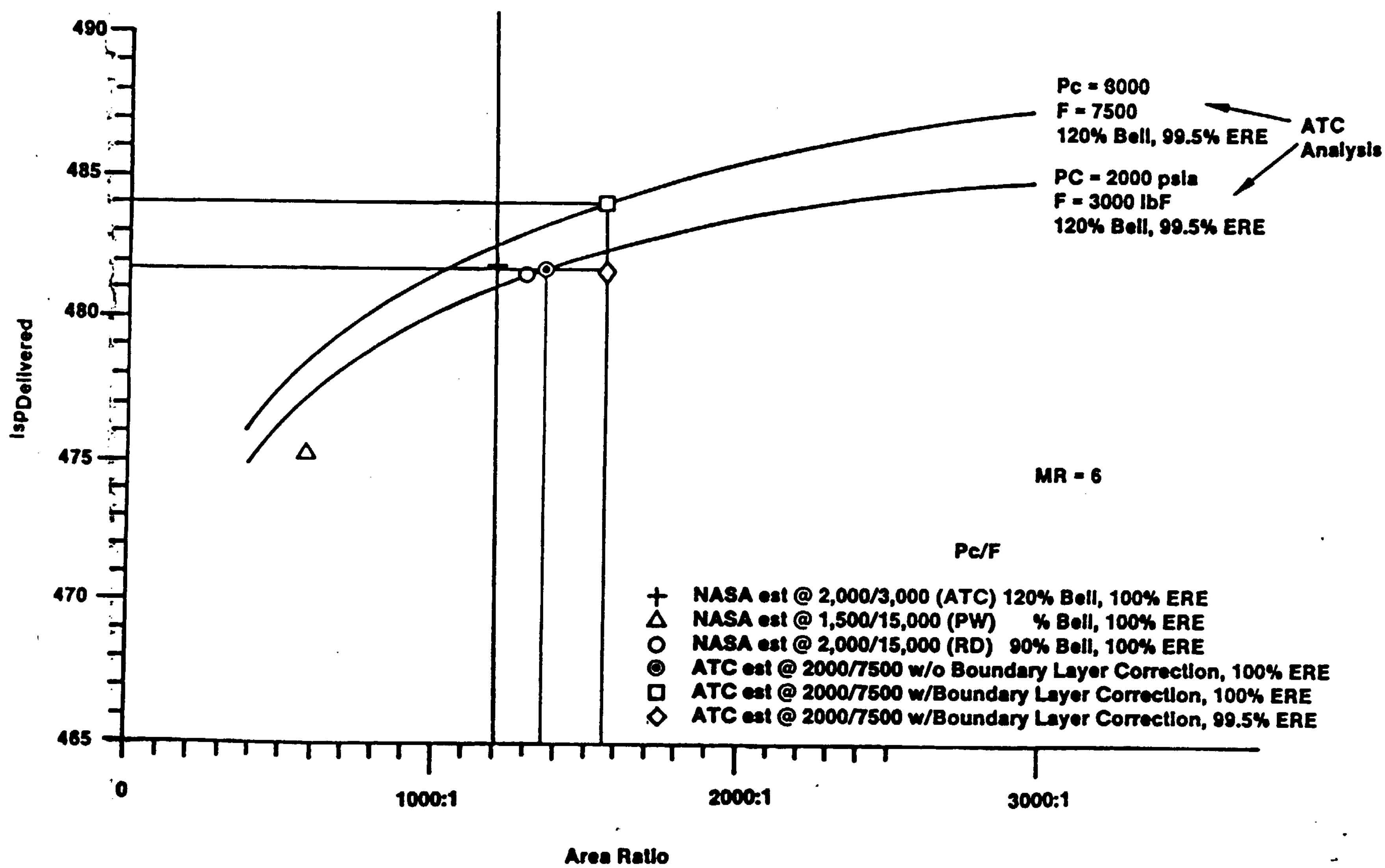


Figure 3.1-13 Predicted LO2/LH2 Engine Performance

3.2 ENGINE OPERATION AND CONTROL

3.2.1 Basic Engine Control Capability

The sequence given in Section 3.2.2 describes the activities related to a normal engine startup, operation at various thrusts, and normal shutdown. A more detailed discussion of operating modes and potential problems is given in Sections 3.2.3 through 3.2.6.

All engine operations are mediated by an electronic engine controller. The controller uses an operating program that incorporates closed loop control features so that all commands are correlated with engine sensor data. This feedback of actual operating data prevents implementation of potential damaging or dangerous commands, and assures safe operation over the entire engine operating range. The engine will not start or continue operation without a functioning controller. The ATC baseline for a controller includes complete redundancy for an initial "fail to operational" capability plus a "fail to limited operation" capability for some additional malfunctions. This would, for instance, allow operation at the last set point with shutdown either on command or at the expiration of the preselected operating time. Any additional failure would cause an immediate shutdown.

The controller selects its program routines based on inputs from either (1) a master operating program, (2) pilot commands input from the flight deck, (3) telemetry commands, or (4) health monitor system commands with safety implications. Under this control scenario the controller, once programmed, will execute the program even if there is a loss of communication with the original programming function. Health monitor system commands, however, must still be processed or the controller will shutdown the engine. One specialized operation, landing on the moon or other body, will require a separate feedback loop to correlate radar altimeter and rate of descent information with engine throttling. This would be a separate program routine with feedback loops similar to those used with the health monitoring system. No completely manual operation of the engine is possible although routines in failed firmware could be bypassed by software in some cases, and manual direction could be given to run certain routines.

Each engine is envisioned as having a stand alone capability so that a failure of one engine does not effect the operation of the other. An exception may be to cross-connect the redundant channels of the engine controller so that either controller could operate both engines. The practicality of this is dependent on hardware design and failure mode analysis. The health management systems are also able to "talk" to each other so that sensor data can be compared and/or reconstructed, but this capability should not be used to have data from one engine shut down another.

3.2.2 Start and Operation Sequence

The following operation description is concerned with the actual operation of valves, igniters, and the interpretation of sensor data. No attempt is made to correlate this with the firmware or software programs as these have yet to be developed. All operations are considered "normal;" no emergency procedures were developed for this report.

3.2.2.1 Pre-Start Operations

(1) Engine Positioning - Doors open, radiation cooled nozzle extended, gimbal operated to position thrust vector through center of mass.

(2) Engine Status Check - Health monitor system powered up, controller functional, vehicle preparations complete for start, engine valves positioned for start (see Table 3.2-1).

(3) Turbopump Chilldown - The turbopumps must be cooled to the same temperature as the propellants to avoid flash evaporation in the pump section with subsequent damaging pump cavitation. In the baseline engine cycle this is done by dump cooling through the pumps with propellant from the tanks. The greater rotating mass of the hydrogen TPA plus its lower operating temperature will require a longer chilldown time than the oxygen system. This start scenario assumes the hydrogen system is chilled first with the chilldown hydrogen allowed to dissipate to space prior to opening the oxygen shutoff valve. The oxygen pump is then chilled down by dumping oxygen until the desired pump temperature is reached. At that time the igniter is turned on, the hydrogen main shutoff valve opened, and the igniter valves opened. This constitutes a tank head start with oxidizer lead. During this chilldown operation the low pressure boost pumps will spin up and the high pressure TPA's will windmill due to the flow of propellant going from tank pressure to vacuum. There will be no usable pump output, and the turbine bypass valves will be in the 100% bypass position.

3.2.2.2 Tank Head Start

The turbopump chilldown is actually completed during the first phase of the tank head start. The start cannot proceed until temperatures are stable at both turbopumps. In the meantime, the low pressure combustion is heating the thrust chamber and nozzle, and hydrogen

TABLE 3.2-1

COMPONENT STATE DURING ENGINE OPERATION

Component	Pre Start	(1) Pump Chilldown	(2) Pump Chilldown	Operation Tank Head Start	Stable Idle	Pumped Idle	Throttle Range	Rated Thrust	Over Thrust	Shutdown
Hydrogen Main Shutoff	Closed	Open	Closed	Open	Open	Open	Open	Open	Open	Closed
Oxygen Main Shutoff	Closed	Closed	Open	Open	Open	Open	Open	Open	Open	Closed
Ox Turbine Bypass	0% BP	0% BP	100% BP	100% BP	100% BP	Modulates	Modulates	10% BP	>0% BP	100% BP
Hydrogen Turbine Bypass	0% BP	100% BP	0% BP	100% BP	100% BP	Modulates	Modulates	10% BP	>0% BP	100% BP
Heat Exchanger	100% BP	100% BP	100% BP	100% BP	Modulates	Modulates	Modulates	Modulates	Modulates	100% BP
Regenerator Bypass	100% BP	100% BP	100% BP	100% BP	Modulates	Modulates	Modulates	10% BP	>0% BP	100% BP
Hydrogen Idle	Open	Open	Open	Open	Modulates	Closed	Closed	Closed	Closed	Closed
Hydrogen Igniter	Closed	Closed	Closed	Open	Open	Closed	Closed	Closed	Closed	Open
Oxygen Igniter	Closed	Closed	Closed	Open	Open	Closed	Closed	Closed	Closed	Open
Hydrogen Back Pressure	Closed	Closed	Closed	Closed	Closed	Closed	Open	Open	Open	Open
Oxygen Back Pressure	Closed	Closed	Closed	Closed	Closed	Closed	Open	Open	Open	Closed
Oxygen Tank Press.	Closed	Closed	Closed	Closed	Closed	Cycles	Cycles	Cycles	Cycles	Closed
Igniters	Off	Off	Off	On	On*	Off	Off	Off	Off	On*
Hydrogen TPA & BP	Stopped	Windmill	Stopped	Windmill	Windmill	Pump Output	Output	Rated Output	Up To Pressure Limit Output	Goes to Stop
Oxygen TPA & BP	Stopped	Stopped	Windmill	Windmill	Windmill	Pump Output	Output	Rated Output		Goes to Stop

*To prevent popping or uncontrolled ignition.

and oxygen flowing through their respective regeneratively cooled sections are gaining usable enthalpy. The hydrogen idle valve is modulating to control mixture ratio. Chamber pressure climbs to a few tenths of a psia until it is in equilibrium with the pressure drops from the tanks to the chamber. A stable idle point should be reached where the engine operates at pressure and thermal equilibrium.

3.2.2.3 Pumped Idle Operation

The turbine bypass valves are slowly closed as the hydrogen idle valve is closed to divert hydrogen flow through the TPA turbines. Turbopump rotation increases with modulation of the bypass valves to keep mixture ratio within limits. Pumped idle is defined as a stable operating point where pump output exceeds tank pressure, but is below the normal throttle range. It is hardware dependent and cannot be predicted at this stage of the design. The transition from tank head idle to pumped idle is a critical operation as is the transition beyond pumped idle to the normal throttle range. That range is defined as a chamber pressure of 200 psia to 2000 psia for 10:1 throttling. The pumps must "bootstrap" from tank head idle to any point in this range. The "bootstrap" capability is given by the low pressure turbopumps.

3.2.2.4 Normal Throttle Range

The turbine bypass valves will be flow tested and models developed for operating characteristics so that they can rapidly move to a predicted state point with only minimal modulation once that point is reached. Both valves would move in unison to a new thrust value but the hydrogen valve would modulate about the setting to adjust the propellant mixture ratio. This is computed from flowmeter data with a backup check of propellant pressures entering the manifolds of the injector. Valve set points would be adjusted after the predicted and actual values are compared. To prevent hunting for a stable set point there will be a bandwidth for an acceptable flowrate for the commanded operation.

The regenerator and heat exchanger bypass valves modulate to control propellant temperatures during operation. Control of the propellant temperatures is required to ensure safe, compatible operation on the basis of material limitations. Material studies conducted for LOX operation have been conducted safely with LOX up to 400°F. This has been selected as a temperature limit for the LOX circuit. For copper alloys, as assumed for the baffle analysis, material properties limits gas-side wall temperature to 1000°F. For cooling with hydrogen, this corresponds to a hydrogen bulk temperature of 900°F.

Based on the limitations of the materials assumed in the analysis of this engine, the valves are designed to regulate flow and temperatures within the above described regions. The HEX valve regulates the hydrogen flow to regulate the oxygen temperature. Margin must be ensured since the oxygen is heated by both the HEX and the nozzle. Exiting oxygen from the nozzle is limited to 400°F which has been demonstrated as a safe operating point for the materials used in the LOX TPA as shown in Refernce 1.

The limit on the hydrogen temperature is set by the materials used in the baffle plate construction. For long life, the desired maximum wall temperature for continuous operation is 900°F. Short term excursions to 1000°F, do not severely limit the cycle life of the component. These temperature excursions are required during very low level thrust (throttle condition) operation and during overthrust operation (greater than 7500 lbf).

As noted in the discussion of the engine power balance, the use of a regenerator allows for a wider throttling range, but a penalty is the increase in chamber wall temperature as the regenerator heats the hydrogen entering the regen chamber. The coolest operating point and, therefore, the longest life point is at 5625 lbf thrust. Highest temperatures are encountered below 500 psia chamber pressure (1875 lbf thrust). Prolonged throttled operation below this point will reduce engine life. Optimum life operation is in the range of 500 psia to 2000 psia chamber pressure. Time restrictions should be established for operation above and below these pressures.

A controls study indicated that at rated thrust a 10% change in throttle setting could be accomplished in about 0.35 second. The entire throttling range was not mapped, and it is unlikely that this would be true at lower thrust levels due to the TPA operating characteristics and the entrance into a region where the oxygen phase change takes place due to 2-phase flow and rapidly changing properties are encountered. The control system needs to work with stable temperature and pressure readings. Slower throttle response is expected below 500 psia chamber pressure due to these factors. Under overthrust conditions throttle-up commands will be limited by programmed control valve operation to reduce temperatures. When these limits are encountered only throttle-down commands would be accepted.

3.2.2.5 Overthrust Operation

Operation above 7500 lbf thrust (>2000 psia chamber pressure) is possible for short times at mixture ratios above six. The entire envelope was not mapped, but one operating point of 2350 psia and MR = 7 was found. This is about 8800 lbf thrust. The overthrust range would be usable for emergencies or other short duration requirements at some penalty to engine life. To use it would require the turbine bypass valve to close to less than 10% bypass. As thrust is increased above 7500 lbf the percent bypass decreases along with the control range. At zero percent bypass, thrust would have to be modulated to keep mixture ratio within limits as the turbopumps are essentially at full power and the only change possible is to increase bypass and reduce thrust. This is an interesting controls problem whose practicality hasn't been assessed.

3.2.2.6 Engine Shutdown

Shutdown is straightforward but somewhat lengthy due to the large residual propellant volume below the main shutoff valves. During detail design when propellant residual volumes for the system are calculated a valve lead or lag time will be determined to give the smoothest shutdown transient. For this preliminary design the assumption is that both main shutoff valves can be closed simultaneously. At the same time the turbine bypass valves are commanded to 100% bypass as are the regenerator and heat exchanger bypass valves. To avoid "popping" and uneven combustion during the shutdown the igniter circuit is activated and the igniter valves opened. A shutdown transient is expected to be in the two to three second range depending on throttle position when the shutdown command is given. Residual propellants should vaporize and disperse in space as all flow passages in the engine below the shutoff valves connect to the thrust chamber.

3.2.3 Tank Head Start

Requirements for the OTV engine include a capability for successful starts at tank pressures equivalent to the vapor pressure of hydrogen at 37.8°R and oxygen at 162.7°R. This is approximately 20 psia and 15 psia, respectively. On start the engine chamber is effectively at vacuum. The unaided maximum pressure differential available for start, then, is 15 psia which is set by the oxygen circuit. All line and component pressure drops must be accommodated within this pressure differential less a small combustion chamber pressure. This is a very small driving force for a rocket engine and poses a number of problems which are discussed below.

3.2.3.1 Pressure Fluctuations from Propellant Boiling

The engines are very likely to be at temperatures approximately 200 to 300°R higher than the entering propellant. To minimize propellant boiling, lines, turbopumps, and valves must be chilled to the same temperature as the propellant. During this chilldown period, some erratic operation of flowrates and pressures will be encountered. A start at this time could be characterized by rapid chamber pressure fluctuations, uncontrollable mixture ratio changes, and intermittent loss of flame with subsequent reignitions and pressure spikes. Such operation must be avoided. One means is to dump a specific amount of the propellants through the engine without ignition. This can cost several hundred pounds of propellant to achieve chilldown. Another option is to recirculate the chilldown propellants back to the tanks through recirculation loops. For such loops to work, however, there must be a recirculation pump in each circuit. The obvious choice is a small electrically driven pump located near the propellant tank outlet. It does not have to have a flow capacity of more than 10% of the full-thrust flowrate. With such a pump chilldown will be rapidly completed with little or no loss in propellant. A bonus is rapid tank pressurization from the returning propellant due to heat pickup from the engine and propellant conversion to the gas phase in the tanks.

The recirculation loop also requires added engine valves and lines to prevent the propellant from entering the injector during chilldown. The recirculation pumps can be continued in operation during the start to add a very useful boost in pressure to the engine. Once pumped idle is reached so that the autogenous pressurization system is operational they would be turned off.

A compromise start/chilldown procedure would use some initial propellant dumping with ignition started just as soon as sensor data indicated stable pressures and temperatures at the turbopumps.

3.2.3.2 Susceptibility to Combustion Transients

Combustion adds additional pressure fluctuations that, at low feed pressures, can be coupled to the feed system dynamics as a "chug" instability. This can be severe enough to damage equipment or couple with the vehicle dynamics to effect the whole vehicle as a "pogo" instability. The recourse is an immediate shutdown. Avoidance is best done by a rapid acceleration through the potential "chug" range to a stable engine operating point. We do not expect a prolonged period of operation at tank head idle for this engine; the available pressure drops are not high enough to preclude "chug" despite the expected smooth combustion with the proposed injector elements.

3.2.3.3 Intermittent Combustion

The very low tank pressure requires a very low combustion chamber pressure for sustained operation. One concern is that the pressure spikes on lightoff may be high enough to stop propellant flow from one of the circuits. At that time the flame would extinguish only to be followed by another lightoff and pressure spike. This would be evident as a rapid "popping" that could be of increasing or decreasing magnitude. Continuous ignition during start would lessen the possibility of "popping," but not entirely eliminate it. A better solution is to increase the system pressure to a value above the lightoff pressure spike.

3.2.3.4 Lack of Repeatability from Start-to-Start

A highly repeatable start transient is very useful in mission planning. With a tank head start there will be some unavoidable variability between starts. This can be avoided by compressing the time between lightoff and reaching a stable operating point. Prolonged operation at tank head idle would be characterized by large variations in delivered impulse for a given operating time. Such variations may be acceptable if they can be calculated in real-time.

3.2.3.5 Control Elements for Tank Head Start

A successful tank head start will require low pressure drop flow circuits. The fuel idle valve opens on start to bypass the hydrogen turbopump circuit for a low pressure drop propellant route to the injector. As combustion stabilizes and more propellant flows through the fuel TPA circuit the fuel idle valve can be closed. Mixture ratio is controlled by the modulation of the variable position fuel idle valve.

3.2.3.6 Two Engine Thrust Vector Control

With side-by-side mounting of two OTV engines the vehicle center of mass should be on a line between the two engines. Any difference in thrust between the two engines during start must be corrected by a gimbal movement of one or both engines or by firing of an attitude control thruster. Smoother operation will result if the start transient on both engines is very nearly identical and characterized by a rapid increase to a stable operating point.

3.2.4 Pumped Idle Mode

Pumped idle is defined as engine operation where the turbine power is sufficient to overcome the tare torque of the TPAs. It represents minimum thrust operation with active turbomachinery. Up to that time the engine operating temperature and chamber pressure are relatively low since chamber pressure cannot exceed tank pressure minus system pressure drops. When the TPAs begin to rotate the propellant pressure in the engine system is no longer limited by tank pressure. As an operating point, pumped idle would be somewhat above the start rotation condition so that system fluctuations would not stall the pumps.

3.2.4.1 Control in Pumped Idle

At pumped idle the fuel idle valve is closed, and all propellant flows through the TPA circuits. The pressure operated back-pressure valves are functioning as high pressure drop flow restrictors (they never completely shut off flow and open completely when chamber pressure reaches 500 psia ±50 psia). The hydrogen regenerator bypass valve will modulate near 90% bypass (a margin of 5% is required for control authority) to minimize bulk temperature rise and chamber maximum wall temperature. The turbine bypass valves will be very nearly in the maximum bypass position but modulating for thrust and mixture ratio control. The LO_2/GH_2 HEX bypass valve will modulate to keep the GOX turbine inlet temperature below 400°F.

3.2.4.2 Stability

The stability of this mode is highly dependent on control range (i.e., valve position range and flowmeter accuracy) and the tare torque differences between the two mechanically separate turbopumps. The two pump's output must be closely matched at all times; otherwise, there is a risk that the rapidly increasing output from the more rapidly accelerating TPA will cause a mixture ratio change beyond the control of the bypass valves. This mode is very difficult to model accurately without actual pump tare torque and performance data.

3.2.5 Engine Throttling and Overthrust Operation

The 10:1 throttling requirement is considered as a capability for fully controlled engine operation from 750 lbf thrust to 7500 lbf thrust at some mixture ratio between 5.0 and 7.0. There is no stated requirement for a specific rate-change in thrust although this is an important consideration in the controls design and there will be a practical rate limit. Relatively slow changes assure close control of mixture ratio.

3.2.5.1 Throttling Applications

There are three likely mission scenarios where throttling is needed or useful:

1) Landing a vehicle on the moon, Mars, or other body within the solar system.

2) A gradual reduction in thrust during propellant depletion to control G-loads on payloads sensitive to acceleration forces.

3) Operation at lower than rated thrust to accomplish a precise orbit adjustment maneuver or rendezvous.

The first application will be the most demanding. If the engine control system can respond adequately to the moon lander mission needs, the other missions will be within the design envelope.

3.2.5.2 Throttling Control

Throttling is done primarily by commanding the oxygen turbine bypass valve to a specific position with the hydrogen bypass valve adjusting as needed to keep within the mixture ratio range. During throttle down operation the hydrogen regenerator bypass valve will allow more bypass due to the hydrogen bulk temperature rise through the chamber. The LO_2/GH_2 HEX bypass valve would also increase the hydrogen bypass based on the bulk temperature rise in the oxygen cooled nozzle. The principal valves used during throttling, however, are the turbine bypass valves. The hydrogen turbine bypass valve action has been modeled and is given in Figure 3.2-2. The distinctly non-linear valve action reflects the interaction with the hydrogen regenerator bypass valve. At full thrust the regenerator bypass valve is limiting hydrogen temperature exiting the baffle plates to about 900°F, and the turbine bypass valve is only bypassing 6% of the hydrogen so that nearly maximum power can be extracted from the turbine. The regenerator is, in essence, the source of thermal power for thrusts greater than 5125 lbf. Similarly, the waste heat picked up by the oxygen in the LO_2/GH_2 HEX provides about two-thirds of the thermal energy needed to operate the oxygen TPA. It should be noted that the cycle is limited by the maximum hydrogen temperature corresponding to a thrust chamber materials limit, and by an assumed safe operating temperature for oxygen with the oxygen TPA materials. The baseline chamber pressure is attained near the limits for these materials. Any higher normal chamber pressure is dependent on a selection of materials with better high temperature strength and oxygen compatibility.

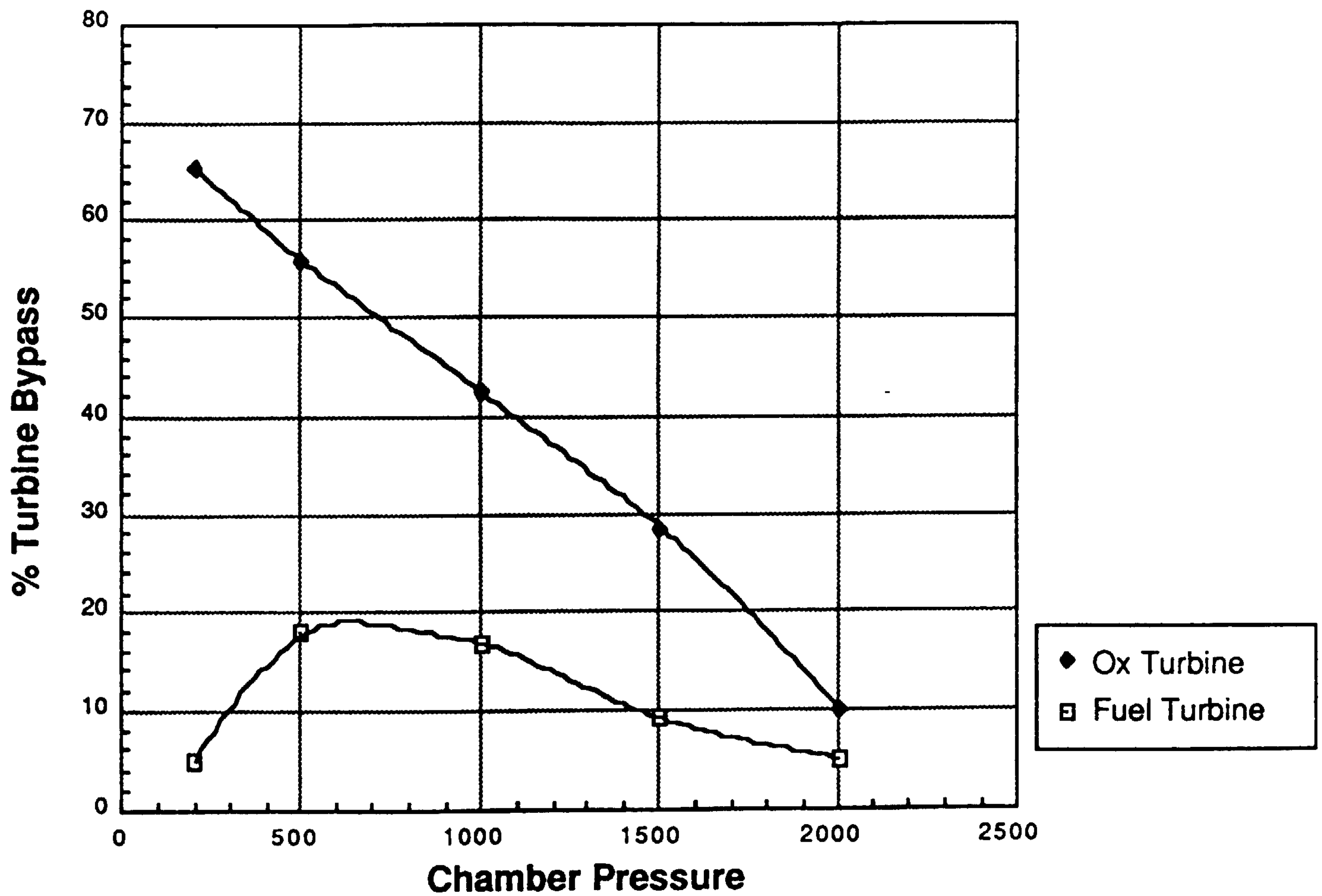


Figure 3.2-2. Turbine Bypass for OTV Engine
(At MR = 6)

Overthrust operation is possible as long as there is any control authority for the turbine bypass valves and maximum temperatures are not exceeded. For emergency operations 7500 lbf thrust can be exceeded at mixture ratios greater than six. This may be needed for lander missions, but should be considered a backup capability available with the basic engine design. It is not attained without a penalty of reduced engine life.

3.2.6 Purge Gas and Power Requirements

3.2.6.1 Purge Gas and Pneumatic Requirements

Purge gases are used in rocket engines to prevent the accumulation of propellants in spaces where they could either freeze or react explosively. Also, some oxidizers may react with the ambient environment to form corrosive solutions that could damage the engine. Where these effects are possible, good design dictates use of a purge gas which is non-reactive with the propellants. This gas is usually nitrogen, argon, or helium. Helium is commonly chosen as being inert, of low-density, and noncondensable when used with cryogenic propellants. Helium requires heavy tanks for high pressure storage and pressure regulators and valves. Lines must be plumbed to all the areas requiring a purge. such an assembly is a significant part (20 to 40%) of a propulsion system's weight. It is very important for a space based vehicle such as the OTV to eliminate or reduce the purge system size and weight. The importance of this is worth emphasizing as it makes one of the strongest arguments for the dual expander cycle engine. In any comparison with other candidate engine concepts the need or lack of need for a purge system must be considered when weights are presented. A simple expander cycle engine uses an inert gas purge system and that weight becomes part of the overall engine's system weight. A purge is needed as the oxygen turbopump is driven by hot hydrogen gas and safety dictates that an inert gas interpropellant seal be used. A gear drive system is not an acceptable solution for long life and has seal problems that also require an inert gas purge.

The design approach chosen for the dual expander engine is to eliminate an inert gas purge completely. This is possible given space based operation and an oxygen turbopump driven by 400°F oxygen. No interpropellant seals are required for either turbopump. The engine is self-purging as the ambient pressure is effectively zero and the propellants will flash to gases and quickly dissipate; their low freezing points make freezing inside the engine very unlikely. Start is done with propellants at tank pressure (if possible) rather than a helium start system. At this stage in the dual expander engine design there is no compelling reason to include a purge system.

A pneumatic system would be required for some valve operations and, possibly, for engine gimballing. As this adds additional servicing requirements and more failure modes, one of the design goals was to perform these functions electrically. The results of the valve and gimbal design tasks favored an all electrical actuation system, so there was no need for a pneumatic system.

3.2.6.2 Electrical power Requirements

The dual expander engine requires electrical power for the following operations/functions:

1) Valve operation
2) Gimbal actuation
3) Engine controller operation
4) Health monitor sensors and signal conditioning
5) Radiation cooled nozzle extension/retraction
6) Component thermal control (heaters)
7) Cool down circulation pumps

Electrical power requirements will vary with the engine status in the operation cycle. Fourteen operating points were selected as representative of the spectrum of power usage. They are defined in Table 3.2-2, The estimated power (watts) for each point are matrixed in Table 3.2-3. Steady state operating power can vary from 498 to 784 watts per engine. Assuming two operating engines and a propellant load of 155,000 lbm with 90% used by the main engines, the per mission operating time would be one hour and 15 minutes. The watt hours would be approximately 623 to 980 per engine or 1246 to 1960 per vehicle. This would have to be increased for components needing continual thermal conditioning by heater circuits.

Two high wattage items are the main propellant shutoff valves. It is assumed that these valves are powered open and spring loaded closed. These valves would require constant current flow to stay open. This was estimated at 97 watts each. This high electrical use would require some heat dissipation capability to be incorporated into the valve otherwise it would be life limited or would be heating the cryogenic propellant. A lower power alternative would be to design the valve with a solenoid latch open feature. A loss of power would de-energize the solenoid, release the latch, and allow the spring to close the valve. The result would be a continuous power consumption when open of about 5 watts instead of 97 watts.

TABLE 3.2-2

ENGINE OPERATION SEQUENCE

Engine Operating Point

STATE	
00	• Nozzle extension or retraction
0	• Engine out gimballing of operating engine
1	• Close fuel and ox turbine bypass valves (pump chilldown) Open fuel regen bypass valve Open fuel igniter valve Open fuel idle valve
2	• Open fuel main valve. Gaseous H_2 flows through pumps, nozzle regen jacket, baffles, idle valve and into chamber
3	• Activate spark igniters. Open ox main valve
4	• Open ox igniter valve
5	• Open ox turbine bypass valve. Gaseous O2 flows through pumps, heat exchanger, nozzle cooling jacket, turbine bypass and into chamber. Thrust chamber temperature is monitored to verify ignition
6	• Modulate fuel idle valve to control mixture ratio

Start Up to Full Thrust

7	• Monitor fuel bypass flow temperature. As temperature rises, modulate fuel idle valve to closure
8	• Turbine speed monitored to determine start of rotation, bypass valves used to control mixture ratio
	• Concern that one TPA output increases faster than the other exceeding mixture ratio limit
9	• Start acceleration to operating thrust level
	• Deactivate spark igniters. Close ox and fuel igniter valves. Close fuel idle valve.
10	• Begin main tank autogenous pressurization
11	• Modulate heat exchanger and H_2 regenerator bypass valves to control ox and fuel turbine inlet temperatures
12	• Modulate ox turbine bypass valve and regenerator bypass valve to control thrust level, fuel turbine bypass valve to control mixture ratio

Table 3.2-3 Representative Engine Power Requirements (Watts)

COMPONENT \ OPERATION	00	0	1	2	3	4	5	6	7	8	9	10	11	12
REGEN BYPASS		60										60	60	60
H. TURB. BYPASS		60	60								60	60	60	60
O. TURB. BYPASS		60	60					60	60	60	60	60	60	60
HEX BYPASS		60										60	60	60
H● TANK PRESS.												24	24	24
O● TANK PRESS.												24	24	24
O● MAIN		97					97	97	97	97	97	97	97	97
H● MAIN		97		97	97	97	97	97	97	97	97	97	97	97
H● IDLE			50	50	50	50	50	50						
H● IGNITER			24	24	24	24	24	24	24	24	24			
O● IGNITER					24	24	24	24	24	24	24			
GIMBAL ACT.		385												
NOZZLE EXT	100													
SENSORS	56	56	56	56	56	56	56	56	56	56	56	56	56	56
HEATING	100	100	100	100	100	100	100	100	100	100	100	100	100	100
CONTROLLER LOGIC	50	50	50	50	50	50	50	50	50	50	50	50	50	50
CONTROLLER VALVE DRIVER ELECTRONICS	20	77	39	34	39	39	58	70	60	60	72	96	96	96
TOTAL	326	1102	439	411	440	440	556	628	568	568	640	784	784	784

The figures for sensor power are discussed in Section 3.6.2 and Appendix A-2, respectively.

The baseline for the OTV is power generation from a bank of fuel cells. It is evident that an electric power failure to the engine will shutdown the engine. Any failure of the power link between the fuel cells and the engine will jeopardize the mission and must be repaired before the engine can be restarted. An alternative is to include an emergency power source at the engine that could complete the programmed engine operation and make a normal shutdown despite a failure at the fuel cells or in the power distribution net. This alternative power source could be either a battery kept charged by the fuel cells or an engine driven generator with a small battery. The generator drive would be gas from the hydrogen turbine effluent. this alternative power system may very well be necessary to meet manrating and safety requirements for the vehicle.

3.3 DESIGN INTEGRATION

3.3.1 Component Layout

3.3.1.1 Envelope Requirements

The baseline transfer vehicle (OTV) is conceived as having two axial engines mounted side-by-side. A view of such a mounting arrangement is given in Figure 3.3.1-1 with the appropriate envelope dimensions. Engine height from the vehicle interface to the nozzle exit is about ten feet. In the nozzle stowed configuration the engine height will be 60 inches. The nozzle exit diameter is very near 61 inches. This gives a minimum exit plane width of 11 feet 2 inches assuming a one foot separation between nozzles. Gimbal movement in the same plane would increase this to a maximum envelope dimension of 13 feet assuming a ±6° gimbal. For engine out operation additional engine movement is required. The maximum envelope could go beyond 16 feet depending on the engine out gimbal requirement. The correlation between these dimensions and the opening needed in an aerobrake is highly dependent on clearances, aerobrake thickness, and aerobrake curvature, but it is evident that the opening will be greater than 12 feet across at the widest dimension and 5 feet across at the narrowest. Aerobrake doors are going to be substantial items with a heavy actuation system. This should be considered when evaluating the aerobrake design as a side mounted system may offer some weight advantages. See Figure 3.3.1-2 for the likely configurations.

The design envelope includes a six inch space between the engine/vehicle interface and the top of the injector body. The injector is four inches thick and the distance from injector face to the throat is 10.5 inches. A throat mounted gimbal is assumed. Gimbal dimensions/forces are detailed in Section 3.3.4. Outside diameter of the thrust chamber is about ten inches. The extendible/retractable nozzle limits the component mounting envelope to a maximum of 18.8 inches from the engine centerline. Subtracting the chamber radius of 5 inches from this leaves a packaging envelope of a 13.8 inch thick annulus about 14 inches high for all of the engine components.

3.3.1.2 Component Locations

Detailed layout drawings for the engine are included in Appendix A.4. The two side views with components labeled are given in Figures 3.3.1-3 and 3.3.1-4. Component dimensions derived from the component design subtasks were used for these drawings so that they would be a true representation of the expected packaging.

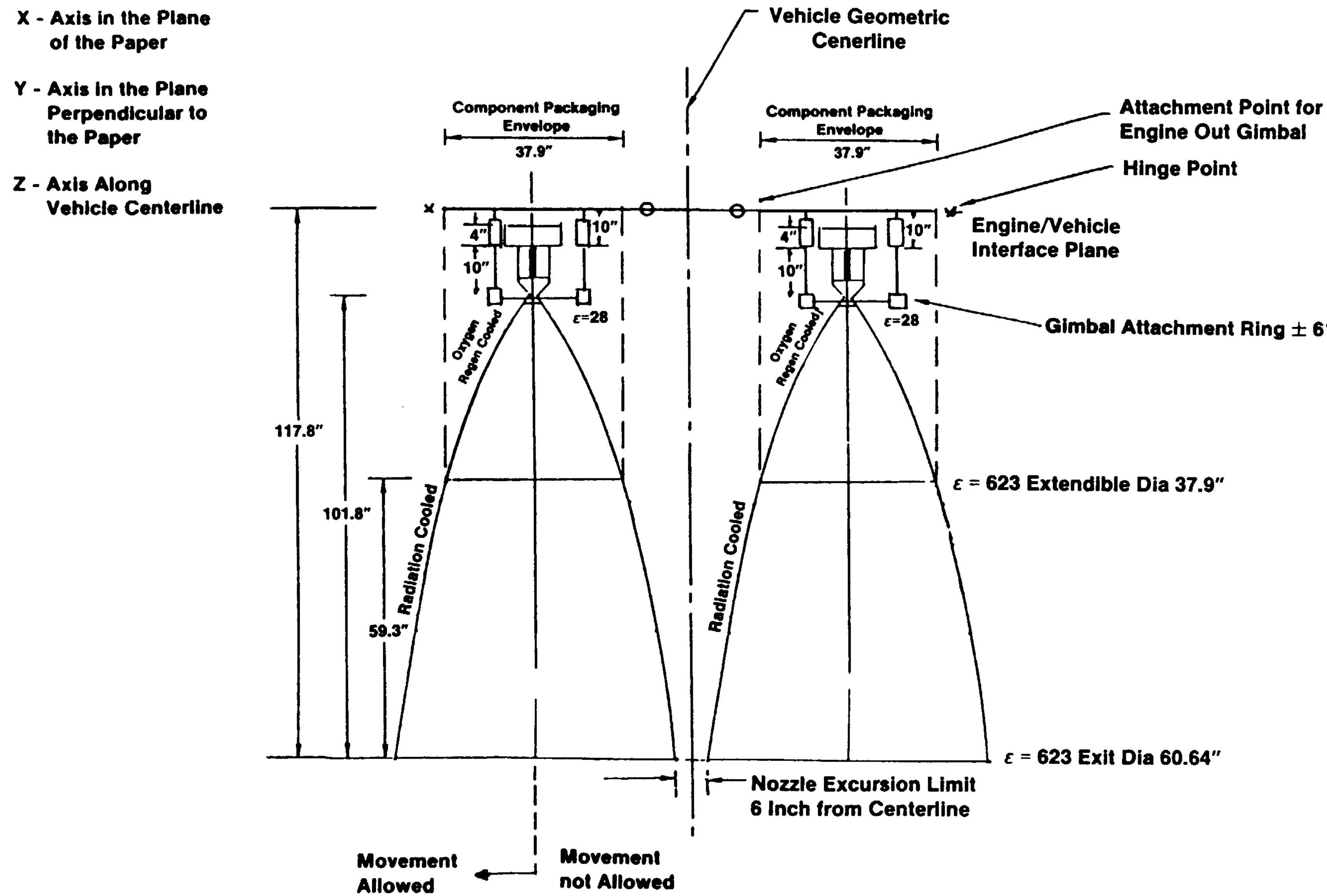


Figure 3.3.1-1 Envelope Requirements Side-By-Side Engine Mounting

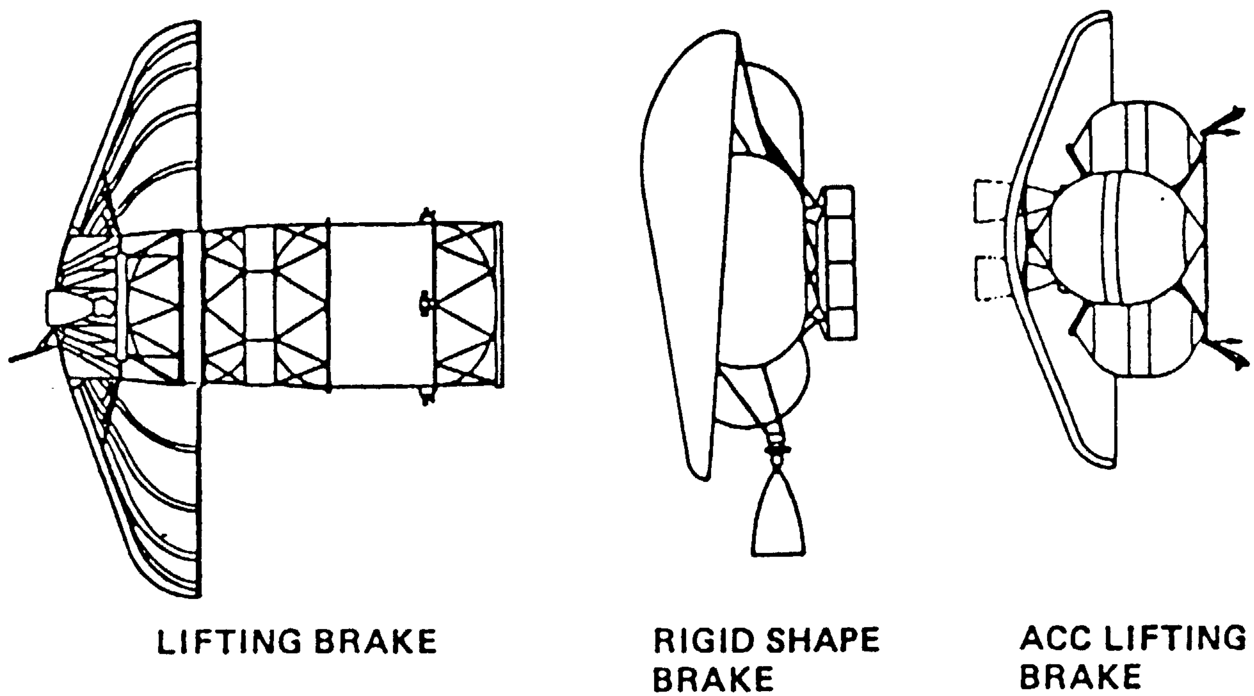


Figure 3.3.1-2 OTV Aerobrake Configurations

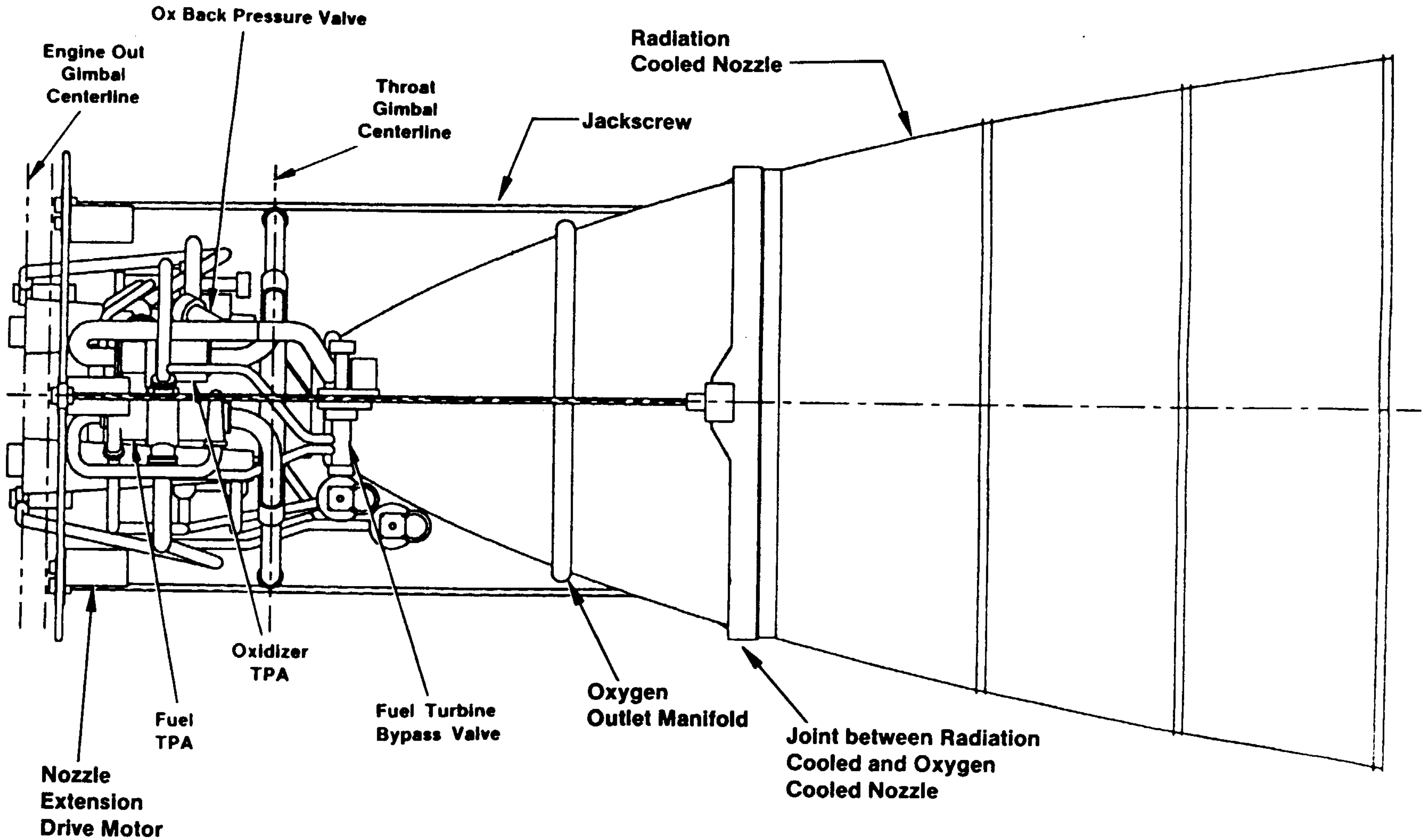
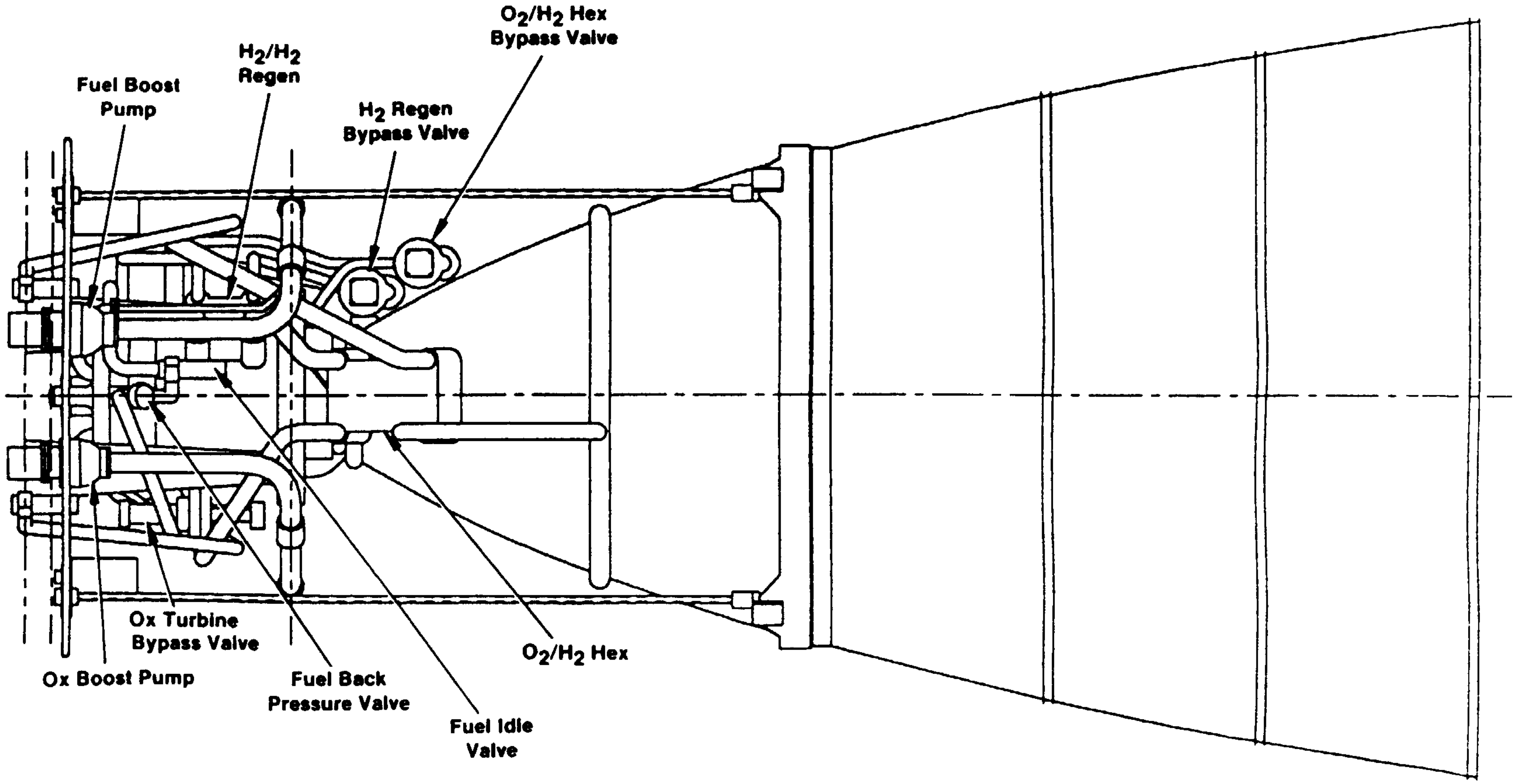


Figure 3.3.1-3 OTV Engine - Turbopump Side



Propellant Inlet Side

Figure 3.3.1-4 OTV Engine Preliminary Design Component Locations

Components are mounted on a metal "can" assembly that is welded to the injector assembly. The other end of the can is attached to the hydrogen inlet manifold via a slip-joint to accommodate thermal expansion. This can assembly also serves to reduce side loads on the engine nozzle. A sketch of the can and gimbal ring configuration is given in Figure 3.3.1-5. The gimbal ring is actually a circular hollow box assembly similar in design to the gimbal used on the Space Shuttle Orbital Maneuvering System (OMS) engines. It is attached to the hydrogen inlet manifold assembly by three brackets.

Thrust takeout assemblies are attached to the engine manifold in two places as shown in Figure 3.3.1-5 and Figure 3.3.1-6. These would be substantial metal structures capable of sustaining the full engine thrust. They connect to the gimbal ring by hinge joints. The ring is located just inside the diameter limit set by the extendible nozzle. The gimbal ring is attached to the engine out gimbal structure by hinge joints and a thrust takeout assembly. This assembly allows the engine to rotate ±6° plus some margin for the allowable movement in the pitch axis. The yaw axis movement is set by the hinge joints between the gimbal ring and the thrust takeout structure from the engine manifold. By locating these two structures 90° apart the engine can move in any direction within the mechanical stops of the hinge joints. The movement is actually done by applying the actuator forces at the head of the engine through a gimbal attachment structure (see Figure 3.3.1-5). Two actuators located 90° apart can provide the full range of movement.

3.3.2 Line Sizes and Pressure Drops

Line lengths were estimated using the component layout drawings. State properties for the oxygen and hydrogen at the flow conditions were called up from the database on the Prime computer, and line sizes were then calculated. Criteria for the size selection were minimum pressure drop and line diameter. These two conflicting criteria required a tradeoff, but line diameters of two inches or less gave acceptable pressure drops in all cases. With the line diameters selected, pressure drops were calculated for all major line segments. They are presented in Table 3.3.2-1 for the hydrogen lines and Table 3.3.2-2 for the oxygen lines. Total line pressure drops of 70 psi for the hydrogen circuit and 50 psi for the oxygen circuit are small in comparison to component pressure drops.

3.3.3 Flex Lines and Joints

The restricted component packaging envelope and in-space maintenance requirements required the plumbing layout design to include several flexible line sections that would normally be hard plumbed. Modern flexible line fabrication makes these lines as reliable as

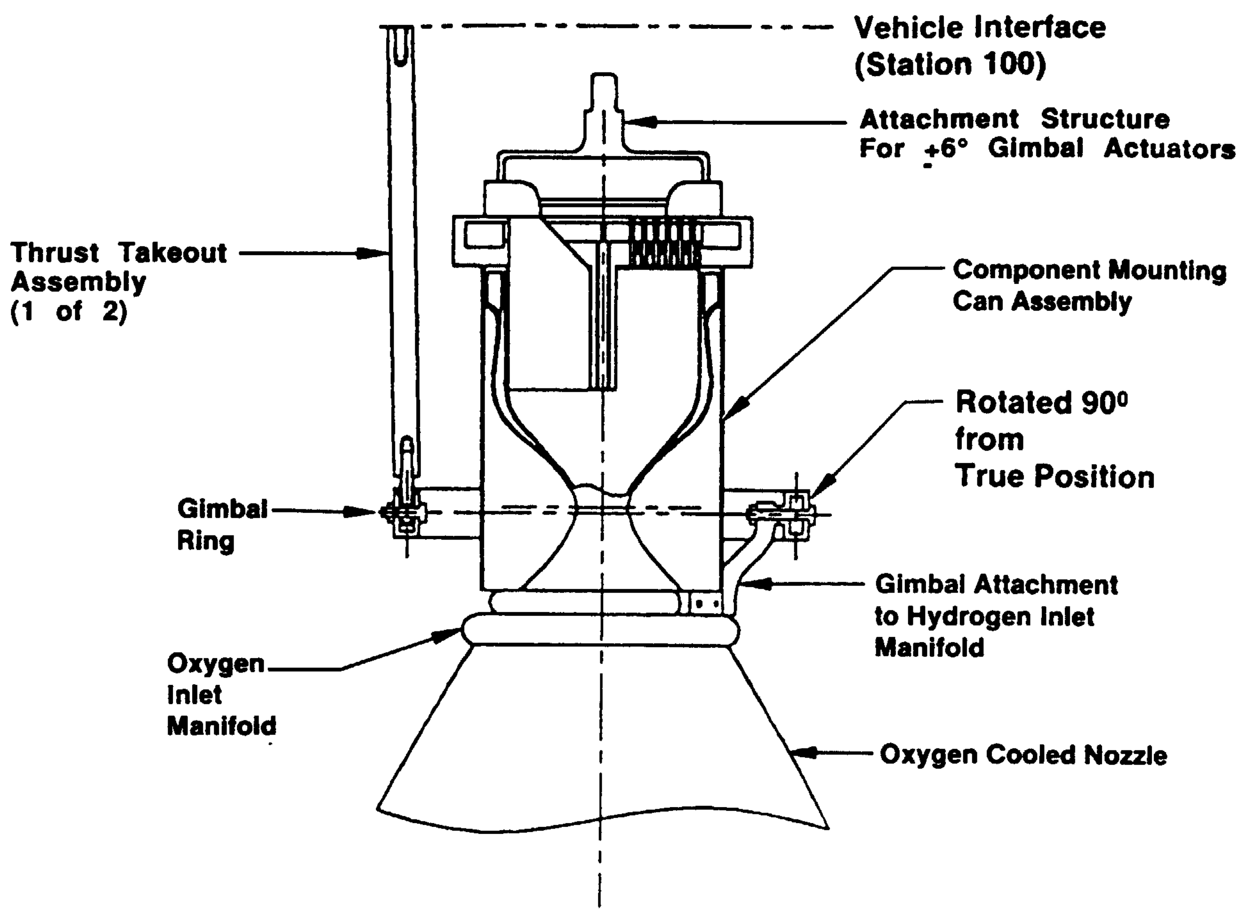


Figure 3.3.1-5 Component Mounting Can Assembly and Gimbal Configuration

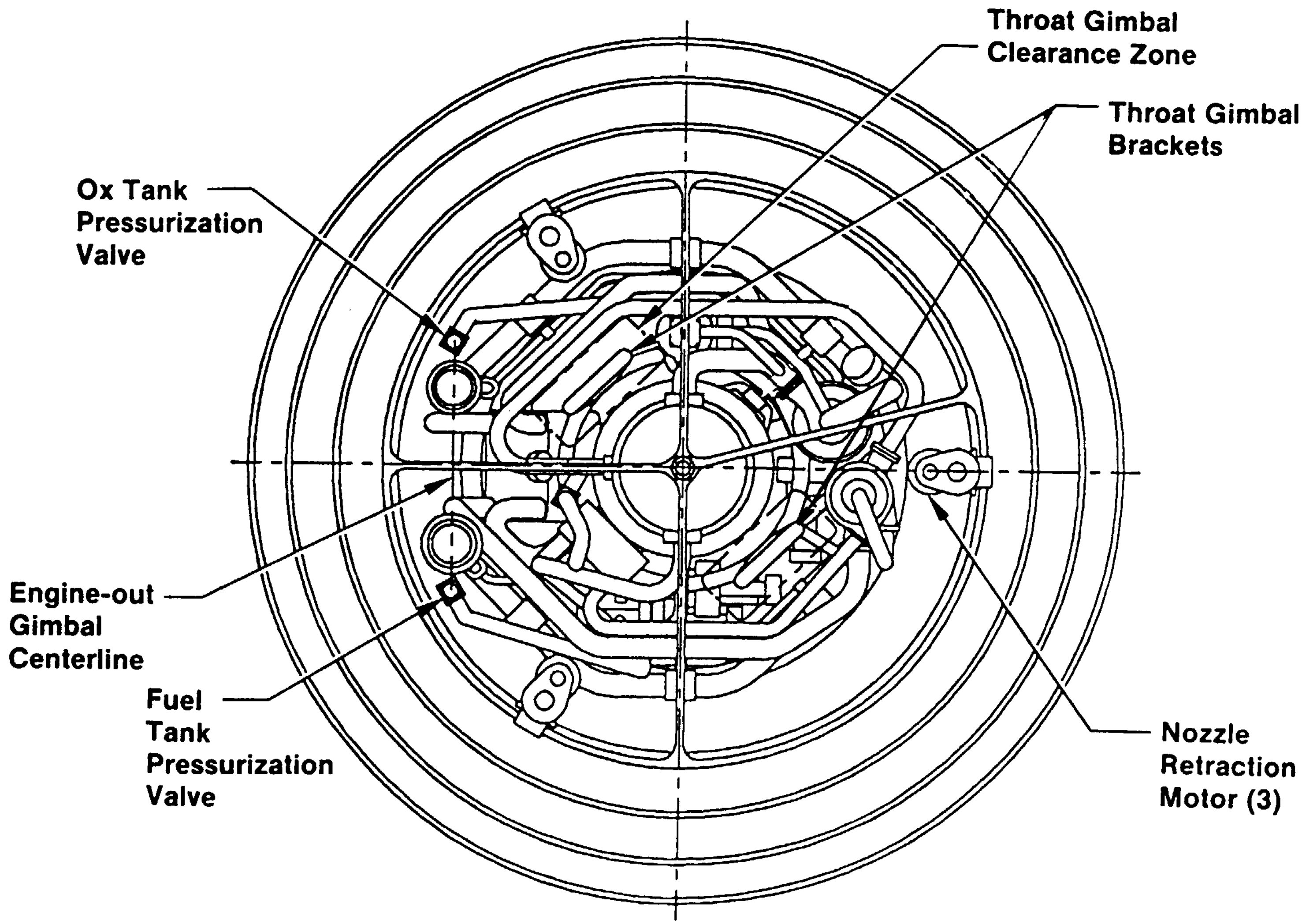


Figure 3.3.1-6 Engine Component and Gimbal Location Top View

Table 3.3.2-1 OTV Concept Design

HYDROGEN LINE SIZING @ MAX FLOW

LINES	OD (IN.)	MAX PRESS psia	FLUID DENSITY lb/ft^3	VELOCITY ft/sec	ESTIMATED ΔP psi
PUMP INLET	2.0	100	4.3	26	0.4
PUMP TO HEX	1.0	6000	4.4	165	14
HEX TO REGEN	1.0	6000	2.8	255	14.5
REGEN TO TURBINE	1.25	6000	0.8	558	17.0
TURB TO HEX	1.5	4000	0.5	542	12.0
HEX TO HEX	1.25	4000	0.9	294	2.00
HEX TO TCA	1.25	4000	1.1	341	10.0
					~70.0
REGEN BYPASS	1.0	6000	2.8	255	30.0
HEX BYPASS	1.5	4000	0.5	542	20.0

<u>APPLICABLE MATERIALS</u>

ALUMINUM

TITANIUM (LOW TEMP ONLY)

AUSTENITIC CRES
- A 286
- NITRONIC 50
- 300 SERIES CRES

INCONEL

HAYNES 188

Table 3.3.2-2 Oxidizer Line Sizing @ Max Flow

OXIDIZER LINE SIZING @ MAX FLOW

LINES	OD (IN.)	MAX PRESS psia	FLUID DENSITY lb/ft^3	FLUID VELOCITY ft/sed	ESTIMATED Δ P psi
PUMP INLET	2.0	100	71	11	1.2
PUMP TO HEX	1.25	6000	70	46	11.0
HEX TO REGEN	1.50	6000	22	104	12.0
REGEN TO TURBINE	1.75	6000	15	206	16.5
TURB TO TCA	1.75	4000	10	144	9.4
					~ 50

APPLICABLE MATERIALS

COPPER LINED AUSTENITIC CRES

MONEL

INCONEL (SOME ALLOYS)

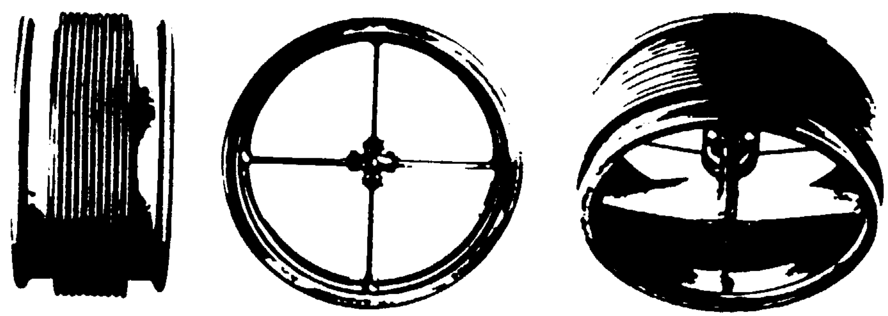
seamless tube lines although inspection is more difficult. The baseline flexible line construction is shown in Figure 3.3.3-1. A stainless metal inner core is covered with a woven metal wire braid. The inner core is designed to accept the operating pressure without the braid, but the braid adds both greater strength and protection for the inner core from rubs and abrasion. Such lines will normally accept 8 to 10 times maximum expected operating pressure (MEOP) before leaking. This is well in excess of the requirements of any design standard. At this stage of the design not all flexible line sections are identified, but they will include the hydrogen and oxygen tank pressurization lines, and the main propellant flow lines from the low pressure boost pumps to the high pressure turbopumps. These four lines cross the engine/vehicle interface and will be subject to movement as the engine is gimballed.

The need for an engine out gimbal requires a significant movement of the operating engine. Propellant inlet lines are routed to the engine at the hinge line for the engine out gimbal. This reduces line movement but requires a significant bend in these two inch lines. A design solution is to use the same type of flex joints that were proven successful in a similar application for the Space Shuttle Main Engine (SSME). This joint design is shown in Figure 3.3.3-2.

3.3.4 Gimbal Concept

The stated gimbal requirement is ±20° in the pitch and yaw axes. This is a very demanding requirement. It would add considerable weight to a gimbal system, significantly compromise the extendible nozzle design due to envelope, and greatly complicate the thrust takeout structure design. After consultation with the NASA Program Monitor it was concluded that the 20° movement was actually required only for engine-out situations. A more modest ±6° movement would be appropriate for normal vehicle operation. This led to the concept of using two separate gimbal systems. The primary gimbal is a lightweight system with limits of ±6° in the pitch and yaw axes. It is assumed that engine movement will not violate the adjacent engine's envelope. A 12 inch clearance between nozzle exits is arbitrarily established as the movement limit. This puts the engine thrust vector 36 inches to the side of the vehicle geometric centerline. With both engines operating this disparity cancels out. With only one engine operating, however, the operating engine must be repositioned to put the engine thrust vector through the vehicle center of mass.

The engine-out gimbal system takes advantage of the symmetry of the system. It only needs to move the engine in one direction. This allows for a single actuator rigidly mounted at the vehicle/engine interface to operate the engine-out gimbal. A mounting platform for the system is shown in Figure 3.3.4-1. The hinge point is towards the outboard side of the platform for

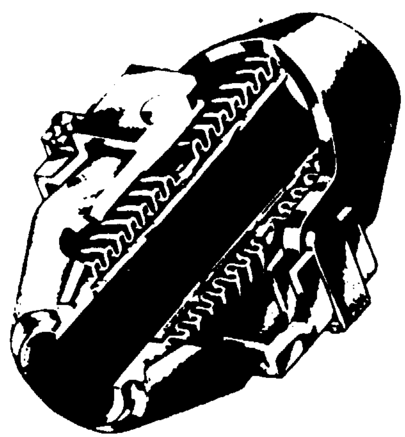


Boost Pump and Turbine
Pump Inlet Lines Will
Contain Flexible Link Joints

Tank Pressurization
Lines Will be Flexible
to Allow Connection
to the Interface

Figure 3.3.3-1 Flexible Lines and Joints

Propellant Inlet Lines May Require SSME Type Flex Joint for 14^0 Platform Gimbal



External Gimbal Ring (Long) Flex Joints

Operating Pressure	4000 to 6000 psia
Temp Range	Amb. to 670^0F
I.D.	.75 to 2.70 Inch Liner
Angular Displacement	$\pm 13.25^0$
Life	200 Operational 1400 Non-Operational Full Deflection Cycles
Material	Inconel 718 Titanium

Typical of

- LPOP Turbine Drive
- Oxid Tank Pressurant
- LPFP Turbine Drive
- LPFP Turbine Disch.

Figure 3.3.3-2 Propellant Inlet Lines May Require SSME Type Flex Joint for 14^0 Platform Gimbal

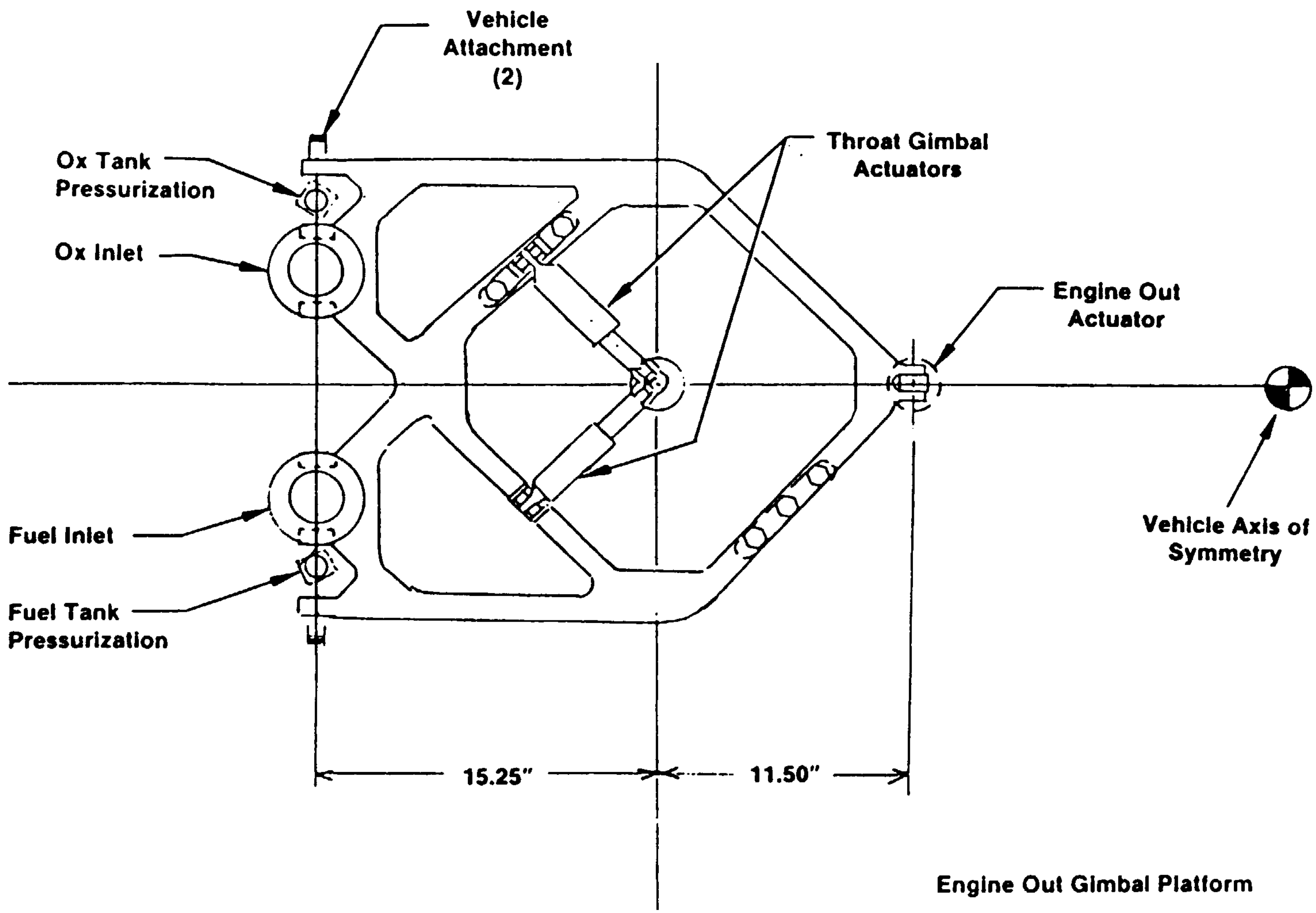


Figure 3.3.4-1 Engine Out Gimbal Platform

each engine. The ±6° gimbal actuators are mounted on this platform. All propellant lines and engine wire harnesses are routed across the interface at the hinge line. To aid in engine removal, all propellant shutoff valves are directly above the separation joints which are hard mounted to the platform. This allows valves to be closed and engine removal to proceed without opening any lines where trapped propellant may be present. The hinge points can also be opened from below so that the complete engine and gimbal platform can be removed as a unit. Thrust takeout is through the two hinge attachments and the engine-out actuator. In the normal fully operational position the platform would be snubbed against the vehicle structure to reduce loads on the actuator and transfer thrust loads through a larger surface. An illustration of the system operation is given as Figure 3.3.4-2.

A summary of program goals and design capability is given in Table 3.3.4-1. With the primary and engine-out gimbal baseline established, calculations were made to determine torques, loads, forces, strokes, and stroke rates. These calculations are summarized in Table 3.3.4-2. Loads and forces are modest and present no serious design problem. The stroke rate for the primary gimbal is fast enough to assure response under all likely operating conditions. Response time may well be limited by the time lags in the system measuring attitude deviations and commanding corrections rather than by the gimbal system response.

The engine-out gimbal response is shown for the 6.67 inch stroke. An engine failure would require immediate operation of the engine-out gimbal as the tailoff impulse from the failed engine would decay to zero within three seconds (estimate). A stroke rate of 1.729 in./sec would allow the operating engine thrust vector to be repositioned 20° within 3.9 seconds. If the correction to place the thrust vector through the vehicle center of mass were near this limit there would be a possibility of a turning moment during the engine-out gimbal movement. This can be compensated for by the use of attitude control thrusters (preferable) or by reducing thrust on the operating engine during the repositioning. The last option is feasible but would be disquieting to crew members who would see one engine fail and the remaining one momentarily decreasing in thrust. Another option available at the cost of higher actuation forces and greater weight is to increase the stroke rate to 2.859 inches/second. This would reposition the engine within the failed engine tailoff impulse.

The system design would have to include several safeguards. The most obvious is an electronic command to the operating engine gimbal to begin movement just as soon as there were engine failure indications in the other engine. These could be main propellant valve closure, an uncommanded decrease in turbopump speed beyond a few percent, or a health monitor

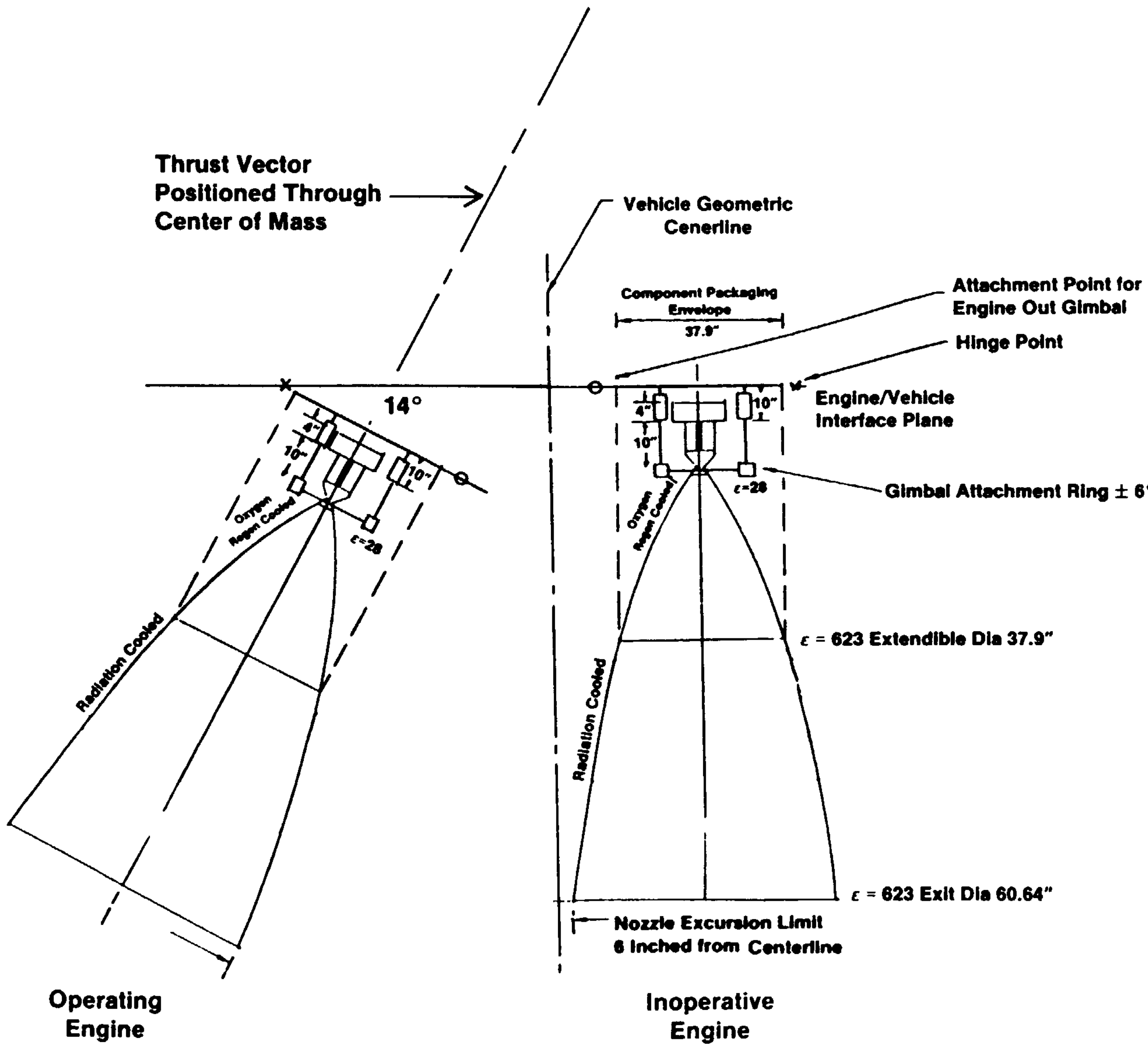


Figure 3.3.4-2 Engine Out Gimbal Position

TABLE 3.3.4-1

COMPARISON OF PROGRAM GOALS AND DESIGN CAPABILITY

	Goal	Design
Gimbal Movement		
Primary, Pitch & Yaw Axes		±6°
Engine Out, yaw Axis		+14°
Total	±20°	+20°, -6°
Envelope		
Engine, Nozzle Stowed Length	60 inch	60.09 inch
Engine Length, Nozzle Extended	120 inch	117.8 inch
Gimbal Type		
Primary	TBD	Throat
Engine Out	TBD	Single Axis Hinge

TABLE 3.3.4-2

DESIGN VALUES

Assumed Requirements		Estimated Torques Breakaway	
Gimbal Velocity	3 deg/sec (max)	FrictionTorque:	
Gimbal Acceleration	60 deg/sec^2 (max)	Primary Gimbal	±4000 in-lbf
		Engine Out	±1500 in-lbf
Pitch Angle	±6°	Thrust Misalignment:	
Yaw Angle	±6°	Primary Gimbal	± 800 in-lbf
Engine	0 to 14°	Engine Out	±115,000 in-lbf

Running with Max Acceleration

Friction Torque:	
Primary Gimbal	±2200 in-lbf
Engine Out	±700 in-lbf
Thrust Misalignment:	
Primary Gimbal	±800 in-lbf
Engine Out	±115,000 in-lbf

Primary Gimbal

- Actuator Load
  - FrictionTorque, m = ± 4000 in-lbf
  - Thrust Misalignment, m = 800 in-lbf
  - Inertia, $\bar{I}$ = 215,100 lbm-in^2 (max)

$$m = \bar{I}\sigma = 215{,}100 \text{ lbm-in2} \times .0027 \text{ rad lbf/lbm-in}$$

$$= 582.8 \text{ in-lbf}$$

Sm = 5383 in-lbf

- Force $= \frac{m}{d} = \frac{5383}{20} = 269.2$ lbf
- Stroke = ±2.102 in (4.204 total)
- Stroke Rate $= \frac{2.102}{6^\circ} = \frac{3^\circ}{\text{sec}} = 1{,}051$ in/sec

TABLE 3.3.4-2 (cont.)

Engine Out Gimbal

- Actuator Load
  - FrictionTorque, m = 1500 in-lbf
  - Thrust Misalignment, m = 115,000 in-lbf
  - Inertia, $\bar{I}$, Assume 50% larger than for main gimbal due to platform and gimbal component inertias
    - m = .5 x 583 in-lbf = 292 in-lbf
    - ΣM = 1500 + 115,000 + 292 = 116,792 in-lbf
- Force = $\frac{m}{d}$ = 4366 lbf
- Stroke = 6.670 inches
- Stroke Rate = $\frac{6.670 \text{ in}}{14°} \times \frac{3°}{\text{sec}} = 1.729$ in/sec
- Alternate Stroke Rate = $\frac{6.670 \text{ in.}}{140} \times \frac{6°}{\text{sec}} = 2.859$ in./sec

  NOTE: This is double the assumed gimbal velocity usual in gimbal design.

system engine shutdown command. There is also a need for a thrust reduction/engine shutdown circuit that would be triggered by an attitude excursion approaching the vehicle's stability limits. The last design option with safety implications would be to add a manual cranking capability to the engine-out actuator. In the event of a failed actuator (not frozen) the system could be cranked to the predicted engine-out setting. This may or may not require an EVA depending on vehicle design.

3.3.5 Weights and Moments

3.3.5.1 Engine Coordinates and Moments

An early set of calculations of component weights and their locations on the layout drawings were used to determine the center of mass and moment of inertia about the center of mass. These calculations are summarized in Table 3.3.4-3 along with the products of inertia. The datum point, station 100.0.0, was selected as a point on the engine/vehicle interface line. This coordinate system is illustrated in Figure 3.3.5-1. Each integer station change is equivalent to one inch.

The engine throat is at station 120.0.0. Engine center of mass is 8.8 inches below the throat and within fractions of an inch of being centered at the intersection of the Y and Z axes. This indicates very good packaging.

The engine component weights were updated towards the end of the design work and the revised figures are presented in the next section. Gimbal calculations and moments were not revised to the new figures as they were close enough to the originals to give little error and available hours were considered better spent on other tasks.

3.3.5.2 Engine Weight Prediction

Engine weights were calculated at two different times during the design. The latest calculation is given in Table 3.3.5-4 along with the center of mass data. All component weights were assumed to be within ±20% of production component weight. When calculating the expected range of total engine weight a simple ±20% gives too wide a variation as it is highly unlikely that all components would come in at minimum weight or all would be at maximum weight. The minimum and maximum weight figures use the square root of the sum of the squares of the differences between nominal and extreme for each component. This statistically based method usually predicts a more realistic overall range for the final weight. The nominal weight of 298.4 lbm does not include several components that could reasonably be assumed to be part of the

Table 3.3.4-3 Mass Properties Summary OTV Gimballed Components

Code	Description	Curr Wt (lbm)	C. of M. from Datum (in.) X	Y	Z	MOI abt C of M (lbm-in2) X-X	Y-Y	Z-Z
1	Thrust Chamber	50.0	125.5	0.0	0.0	5450	21413	21413
2	Ext. Nozzle	97.1	148.0	0.0	0.0	31586	113751	113901
3	OX TPA	9.0	112.0	-9.5	3.3	0	0	0
4	Fuel TPA	10.0	111	-11.5	-1.5	0	0	0
5	OX Boost Pump	6.0	104	15.25	5	0	0	0
6	Fuel Boost Pump	6.0	104	15.25	-5	0	0	0
7	H2/H2 Regenerator	5.5	115	5	-5	0	0	0
8	O2/H2 HEX	22.0	127	9	0	0	0	0
9	Valves	46.2	118.3982	-0.24675	-0.73160	5365.408	6479.868	6778.883
10	Lines	20.0	117	0	0	1300	1300	1300
11	MISC	20.0	117	0	0	1300	1300	1300
Total Sys Depleted		291.8	128.8	0.7	-0.2	52286	205897	212485
Expendables								

Products of Inertia

	X-Z	X-Y	Y-Z
Total System Depleted	-1388	-3438	-630

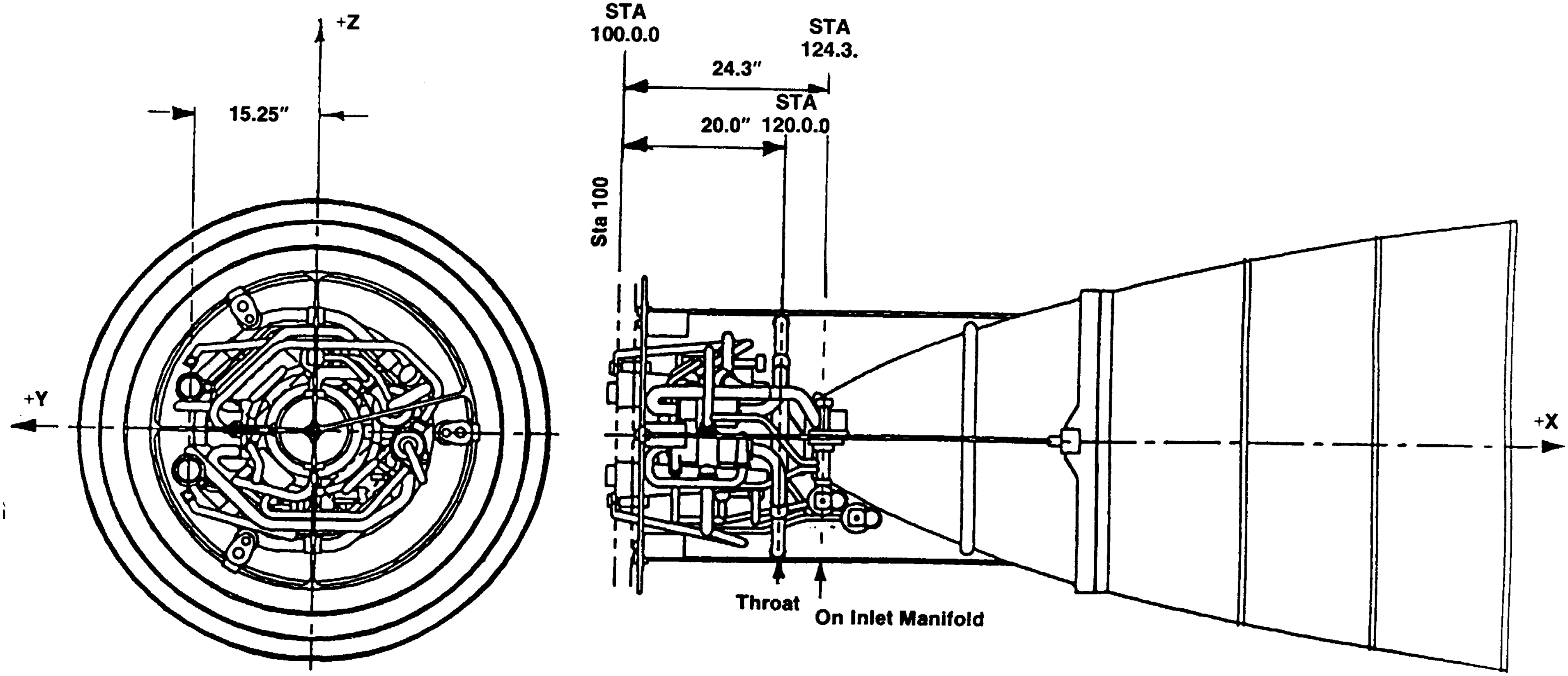


Figure 3.3.5-1 Engine Orientation to Design Datum Point

TABLE 3.3.5-4

PRELIMINARY ENGINE WEIGHT ESTIMATE

	Current Weight			C of M from Datum (inches)		
	(-20%)	Nominal	(+20%)	X	Y	Z
Thrust Chamber	(40.0)	50.0	(60.0)	125.5	0.0	0.0
Ext. Nozzle	(78.6)	98.3	(118.0)	148.0	0.0	0.0
Ox TPA	(7.2)	9.0	(10.8)	112.0	-9.5	3.3
Fuel TPA	(12.0)	15.0	(18.0)	111.0	-11.5	-1.5
Ox Boost Pump	(4.8)	6.0	(7.2)	104.0	15.3	5.0
Fuel Boost Pump	(4.0)	5.0	(6.0)	104.0	15.3	-5.0
H2 Regen	(3.6)	4.5	(5.4)	115.0	5.0	-5.0
O2/H2 HEX	(14.4)	18.0	(21.6)	127.0	9.0	0.0
Valves						
Ox Turb Bypass	(7.4)	9.2	(11.0)	113.0	0.0	9.0
Fuel Turb Bypass	(7.4)	9.2	(11.0)	125.0	-8.0	0.0
Ox Back Press	(4.0)	5.0	(6.0)	114.0	-13.0	6.0
Fuel Back Press	(2.4)	3.0	(3.6)	108.0	9.0	0.0
Ox Tank Press	(2.1)	2.6	(3.1)	101.0	15.3	8.3
Fuel Tank Press	(2.1)	2.6	(3.1)	101.0	15.3	-8.3
Fuel Idle	(4.0)	5.0	(6.0)	112.0	8.0	-2.0
H2 Regen Bypass	(7.4)	9.2	(11.0)	126.0	-2.0	-8.0
Ox/H2 HEX Bypass	(5.4)	6.8	(8.2)	130.0	2.0	-10.0
Valves Subtotal	(42.1)	52.6	(63.1)	118.4	-0.2	-0.7
Lines	(16.0)	20.0	(24.0)	117.0	0.0	0.0
Misc	(16.0)	20.0	(24.0)	117.0	0.0	0.0
TOTAL SYSTEM	274.4	298.1	322.1	128.8	0.7	-0.2

engine. These are components either not gimballed or above the engine/vehicle interface. A preliminary estimate for these components is given in Table 3.3.5-5. Center of mass estimates were not computed for these components as they are attached to vehicle structures.

One assembly not needed for this engine that would add considerable weight is a helium purge system. The baseline design is considered capable of self purging when operating in space. This should be considered when comparing the weight of this engine with other designs. This is a very important discriminator that affects not only total weight but such other factors as servicing, reliability, packaging, and vehicle design.

3.3.6 Materials Selection and Producibility

3.3.6.1 Producibility

A detailed producibility analysis will be completed as part of the detail design phase. For the preliminary design a much more limited analysis was completed. It was subject to the following guidelines and limitations:

- Limited by available hours in the component integration subtasks and low priority.
- Chamber and injector producibility concerns to be addressed by a separate NASA LeRC funded task within the program.
- Heat exchanger and regenerator producibility to be demonstrated in the SSME heat exchanger program.
- Valve and sensor producibility issues given only cursory consideration until there is better component definition.

Four areas were evaluated for producibility. The concerns are summarized in Table 3.3.6-1.

The baseline turbopumps use hydrostatic bearing systems. This is necessary to meet the reliability and service life requirements of the engine. To function efficiently, the bearings must operate with clearances of 0.2 to 0.5 mil over fairly large areas. A larger clearance rapidly translates into severe losses in efficiency as the high pressure fluid for bearing operation is taken from the pump output and discharges into the turbine output. Tolerances for the bearings are

TABLE 3.3.5-5

PRELIMINARY ENGINE WEIGHT ESTIMATE

COMPLETE ENGINE

Component	Current Weight Estimate
Propellant Flowmeters (4)	3.0 lbm
Hydrogen Main Shutoff Valve	8.5
Oxygen Main Shutoff Valve	8.0
Primary Gimbal Actuators (2)	17.0
Engine Out Gimbal Actuator	14.0
Subtotal	50.5 lbm
Nominal Gimballed Component Weight	298.4
TOTAL	348.9 lbm

TABLE 3.3.6-1

PRODUCIBILITY CONCERNS

- TPA Spherical Bearings
  - Few machine shops with competence to meet geometries, clearances
  - Materials unfamiliar to many shops
- Heat Exchanger May Be Made of Inconel or Nickel
  - Limited platelet fabrication experience
  - Bond strength must be demonstrated
  - Selection of bonding aids need to be optimized
- Hot H_2 and Warm O_2 Turbine Bypass Valves Not in Production
  - Limited vendor experience
  - No long life designs
  - Reliability needs to be proven
- Thrust Chamber/Injector Complicated
  - Channel geometry requires many machining steps
  - Flow circuits in injector unusually complex
  - Hydrogen cooled baffles have critical fabrication requirements

correspondingly very tight. This limits the number of machine shops willing and capable of performing the work. During the prefabrication vendor survey for the OTV oxygen TPA there were fewer than a half dozen shops identified in California that were competent to perform the work, and not all would bid on it. The shops with modern numerically controlled machines is increasing, but they also require a competent programmer to function. This is a skill in short supply in most shops. The result is few bidders, high cost, and schedule delays. This situation should be improving during the next few years, but will be a concern for TPA manufacture.

The oxygen TPA also will use various monel alloys and surface coatings of silver or other non-flammable materials. Most shops are comfortable with steel, aluminum, and titanium alloys, the common aerospace materials. The monels are out of context for these shops unless they also do work for the chemical process industry. The lack of familiarity with these materials also leads to no-bids when fabricators are solicited. A solution is to work closely with a small number of selected vendors to develop their capability and assure there will be bidders for these jobs. That requires continued work over the next few years using these materials. Some production work is needed; technology efforts are too short lived and do not lend to repeat business for these specialty vendors.

The baseline construction materials for the LO_2/GH_2 heat exchanger (HEX) is NASA-Z copper. The phase change of the oxygen can be more easily accommodated by reducing the heat transfer coefficient. A means of doing this is to use monel or inconel as the HEX construction material. A deterrent to this is the limited platelet fabrication experience with either of these alloy families. Producibility would need to be proven by developing bonding parameters, verifying bond strength, and selecting bond aids. This could easily require an expensive development program. The alternative is to stay with copper but develop a channel geometry that would promote mixing despite some film boiling. This would also require analysis and some verification testing and could have a cost similar to the monel or inconel platelet development. In other respects the platelet heat exchanger technology is rapidly maturing with the current SSME heat exchanger development program solving many problems.

The turbine bypass valves are hot gas proportioner valves. They function by opening a low pressure drop passage around the TPA turbine in response to engine controller commands. If ideally designed the amount of gas routed through the bypass would be directly proportional to the linear or angular travel of the valve mechanism. This is not a simple design task as flowrates show wide variation, temperatures of entering gas vary several hundred degrees, and the gas pressure can be a few psi or several thousand psi. ATC conducted a vendor survey in

1985 and was unable to find readily adaptable valves from any valve vendor. Development costs were, at that time, estimated at well over $1M. No valve development was funded. In 1988 the Aerojet Solid Propulsion Company (ASPC) was asked to evaluate solid propellant hot gas valves for adaptation to the OTV engine. Engineers from ASPC presented some information on hot gas valves that showed that the OTV operating conditions were well within current state-of-the-art, but reliability for the high number of cycles would have to be proven as solid propellant gas valves have very short cycle life requirements. The current ATC position is that the turbine bypass valves can be adapted from solid propellant hot gas valves but a development program is still needed. Cost and schedule should be acceptable for an OTV engine development program.

The remaining producibility concern was prompted by the complexity of the baffled injector design. The baffle plates are a necessary component in the design as they provide over 60% of the heat transfer area for heating the hydrogen. The baseline design uses a platelet stack to form the flow channels. A cross-section through a channel is shown in Figure 3.3.6-1. This channel is only 0.020 inch wide but 0.20 inch deep. As indicated in the figure, the tip of the baffle is machined to a smooth semi-circle after bonding. Present concerns are to verify the bond strength and the heat transfer through the platelet stack. The bonding will be proven in the SSME heat exchanger program, and a new task started in June 1988 will assess thermal conductivity through a stack of bonded copper platelets.

The regeneratively cooled chamber baselines a microchannel design for optimum life and cooling. Channels are 0.010 inch wide and 0.083 inch deep in the throat with lands 0.011 inch wide. In the 3.0K lbf TCA cooled throat program ATC demonstrated 0.010 inch x 0.040 inch deep channels in NASA-Z material. The new task started in June 1988 will assess the ability to machine the deeper microchannels in GLIDCOP AL-15. The GLIDCOP material was chosen for better cycle life, but machining properties need to be assessed. Fallback would be to the NASA-Z material.

The remaining concern with the thrust chamber involves the complexity of the fabrication. There are many parts, flow paths are complex, and fabrication will involve brazing and several types of welds. A realistic assessment of the producibility will require the design to be completed. At present it is complete only to the preliminary design level.

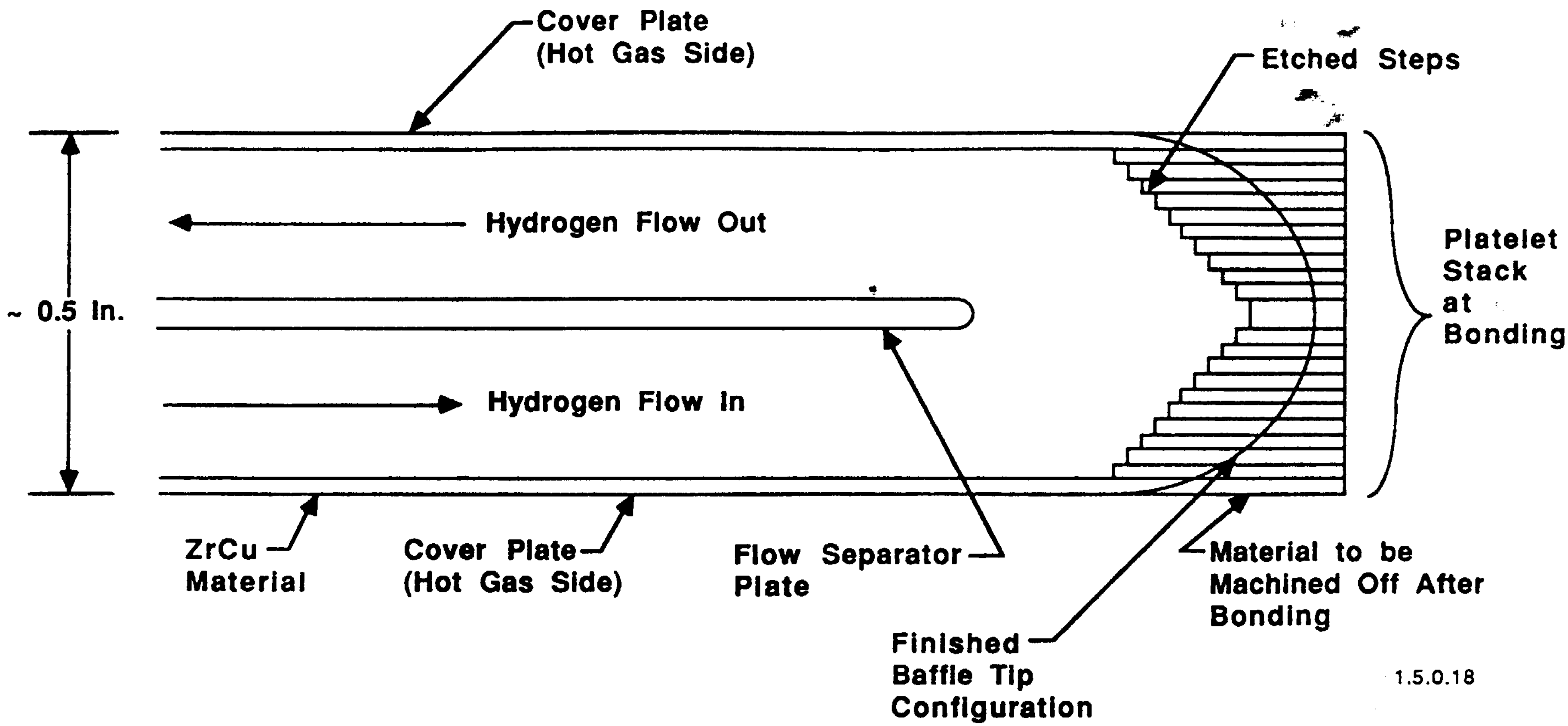


Figure 3.3.6-1 Injector Baffle Plate Fabrication Concept

3.3.6.2 Materials Selection

The criteria for materials selection are:

- Propellant/combustion gas compatibility
- Strength/Weight ratio of candidate materials
- Acceptable properties over the mission temperature range
- Fabricability in the part geometry
- Thermal conductivity for hot section parts
- Compatible coefficients of thermal expansion and galvanic corrosion potential for joined materials
- Data base for material properties
- Susceptibility to radiation induced property changes
- Material availability and vendor reliability

There was no intent to select materials with significant development requirements even though potential benefits were possible. Even the search for 400°F oxygen compatible metals was directed at existing alloys. Also, composite materials were excluded due to the need to verify properties in specific applications and to tailor designs to accommodate preferred lay up or weave fabrication methods. Such materials should be investigated for actual OTV engine development as they can significantly reduce weight for many parts.

The worst case temperature range was assumed to be from 37.8°R (liquid hydrogen temperature) to the component specific temperature listed in Table 3.3.6-2. All materials except the copper alloys have indefinite operation time at these temperatures. Copper alloys are time limited at temperature >800°F as annealing starts above that temperature and they will eventually revert to the annealed properties.

Propellant compatibility was fairly easily assessed. The oxygen compatible materials are well recognized in this industry. Hydrogen compatibility hinged on the susceptibility

TABLE 3.3.6-2

OTV ENGINE COMPONENT THERMAL DESIGN CONSTRAINTS

• Hydrogen TPA Turbine Inlet:	~1250°F Materials in Baseline Design (Titanium, Inconel)
• Oxygen TPA Turbine Inlet:	≤ 400°F Established as a Safe Upper Limit for Oxygen Operation
• Hydrogen TPA Turbine Bypass Valve:	~1250°F for Materials Selected
• Oxygen TPA Turbine Bypass Valve:	≤400°F Safety Considerations for Oxygen Service
• Hydrogen Tank Pressurization Valve:	~1250°F for Materials Selected
• Oxygen Tank Pressurization Valve:	≤400°F for Safety Considerations
• Thrust Chamber Wall	≤800°F
• Baffle Plate Wall	
- Copper Alloy Construction	≤1000°F
- Cu-Ni-Pt Construction	≤2200°F

to hydrogen embrittlement of the material, and, to a lesser extent on hydrogen diffusion rates through the material. The recommended materials are given in Table 3.3.6-3. More detailed lists are given in the TPA and valve discussions for these components.

One material option not pursued in the preliminary design that may be very beneficial from the standpoint of chamber life is the use of a ceramic coating on the copper in the throat. NASA LeRC is funding a program to validate such an approach during FY 88. This could be a very important and relatively simple means of extending chamber life by reducing the copper surface temperature in the very high heat flux environment of the throat. It could also be the means for allowing an increase in chamber pressure above 2000 psia which is at present more a materials limit than a hydrogen heating limit.

3.3.7 Design Areas Not Addressed

The resources available required some limits on the design task scope. Areas not addressed are given in Table 3.3.7-1. None of these items would be expected to present serious design problems. Some, such as the safety analysis, would be best reserved until requirements are better defined and the engine is put in the context of a propulsion system design. It should be noted that insulation requirements could add several pounds to the engine weight and can force a reconsideration of some in-space maintainability options.

TABLE 3.3.6-3

OTV ENGINE MATERIALS SELECTION FOR MAXIMUM PROPELLANT COMPATIBILITY

Component	Initial Selection	Alternates
Thrust Chamber Assembly		
• Injector (Platelet)	Nickel	Incolloy or Monel
• Cooled Baffle Plates	ZR Cu Platelets	NASA-Z/Nickel-Platinum Platelets
• Manifolds	Inconel	Monel for Ox, Inconel for H_2
• Regen Chamber		
- Substrate/Liner	GLIDCOP Al-15	NASA-Z
- Closeout	Nickel-Cobalt Electroform	NiCr or Ni Mn Electroform
• Oxygen Tubing	Monel	Cu-Monel Clad or Cu With Composite Overwrap
• Hydrogen Tubing	Cu-Ni-Ti-Alloy	SS or Inconel
Components		
• Hydrogen TPA (Structure)	Titanium, A-286	None
• Oxygen TPA (Structure)	Monel K-500, K-400	Parts of Nickel, Copper or Silver
• Oxygen Regenerator	ZrCu	Beryllium or Inconel 615
• Nozzle Extension	Coated Columbium	Carbon-Carbon
• Valve Bodies	Monel (O_2), Inconel (H_2)	Non-H_2 Sensitive Steels, Titanium, for H2 Valves

TABLE 3.3.7-1

DESIGN AREAS NOT YET ADDRESSED

- Component and seal redundancy
- Component mounts and bracketry
- Component structural analysis
- Maintainability - welded vs bolted interfaces
- Sensor signal conditioning unit locations and attachments
- Ignition system integration and packaging
- Additional flex joints for lines (misalignment, angularity, thermal, distortion)
- Nozzle and engine stowing
- Insulation
- Safety analysis - manrating

3.4 COMPONENT DESIGN

3.4.1 Hydrogen Turbopump Assemblies

The hydrogen pressurizing system consists of a low pressure boost turbopump followed by a high pressure turbopump, Figure 3.4.1-1. The boost turbopump consists of a single stage pump driven by a four stage hydraulic turbine on a common shaft, Figure 3.4.1-2. The rotor assembly is supported on ball bearings which are cooled by the pumped hydrogen. Labyrinth seals are used between the turbine-bearing-pump as limited leakage can be tolerated when the same fluid is in all components. The boost turbopump is estimated to weigh 5 pounds.

The high pressure hydrogen turbopump integrates two spools consisting of a two stage centrifugal pump driven by a single stage turbine, Figure 3.4.1-3. The pumps are similar except, for the first of the four pump stages, where an axial flow inducer has been incorporated. The turbine rotor of the second high pressure spool, exhausts directly into the first, low pressure spool turbine. The majority of the pump parts are common between the two spools except for the inducer. The turbine nozzles and rotors differ to accommodate the gas density differences between spools. The bearing systems are axially compensated hydrostatic spherical bearings flowing hydrogen. Bearing flow is supplied by each spool's high pressure pump stage. Estimated weight is 15 pounds.

3.4.1.1 Hydrogen Turbopump Design Criteria

Design criteria controlling turbopump design operating conditions are dictated by the OTV engine interface requirements, Table 2-1 and the estimated best obtainable engine specific impulse. The resulting engine flow rates and pressures, tabulated in Table 3.4.1-1, are the basis for the hydrogen turbopump design extreme operating conditions. The basic through flow rates and pump discharge pressures are dictated by the engine thrust and chamber pressure requirements. Suction pressure and temperature come from propellant storage conditions. Turbine flow rate and pressure ratio result from the available turbine inlet pressure and temperature dictated by the thrust chamber hydrogen cooling system heating and pressure drop characteristics.

The relatively low net positive suction head, NPSH, of 15 feet would result in a 20,000 rpm high pressure turbopump with low specific speed pumps and turbine. Low specific speed yields low efficiencies and relatively high weights. As a result, the pumping system selected employs a low speed boost turbopump operating at 20,000 rpm followed by a higher speed high pressure turbopump; a preliminary design specification sheet is given as Table 3.4.1-2.

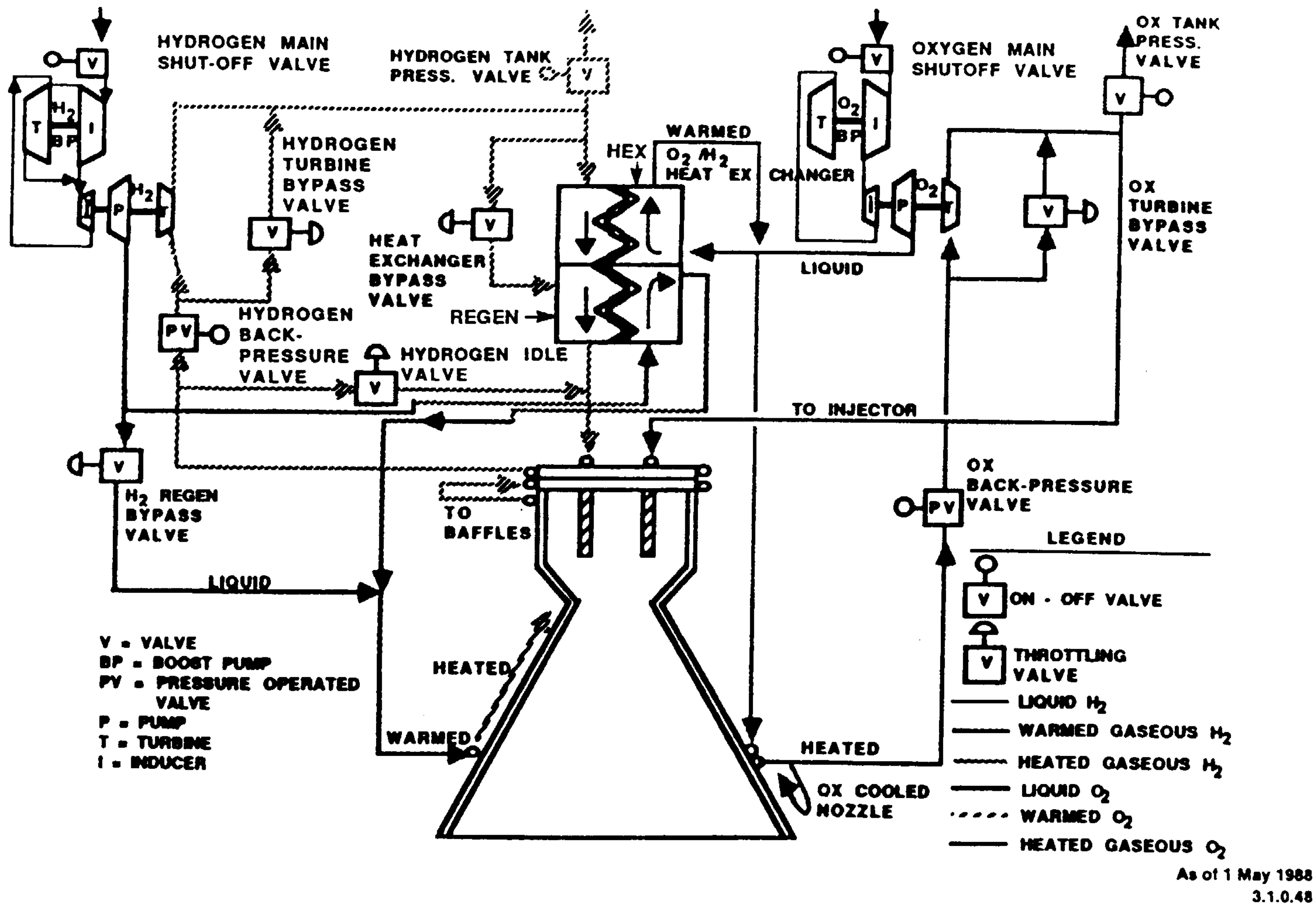


Figure 3.4.1-1 OTV Engine Flow Schematic

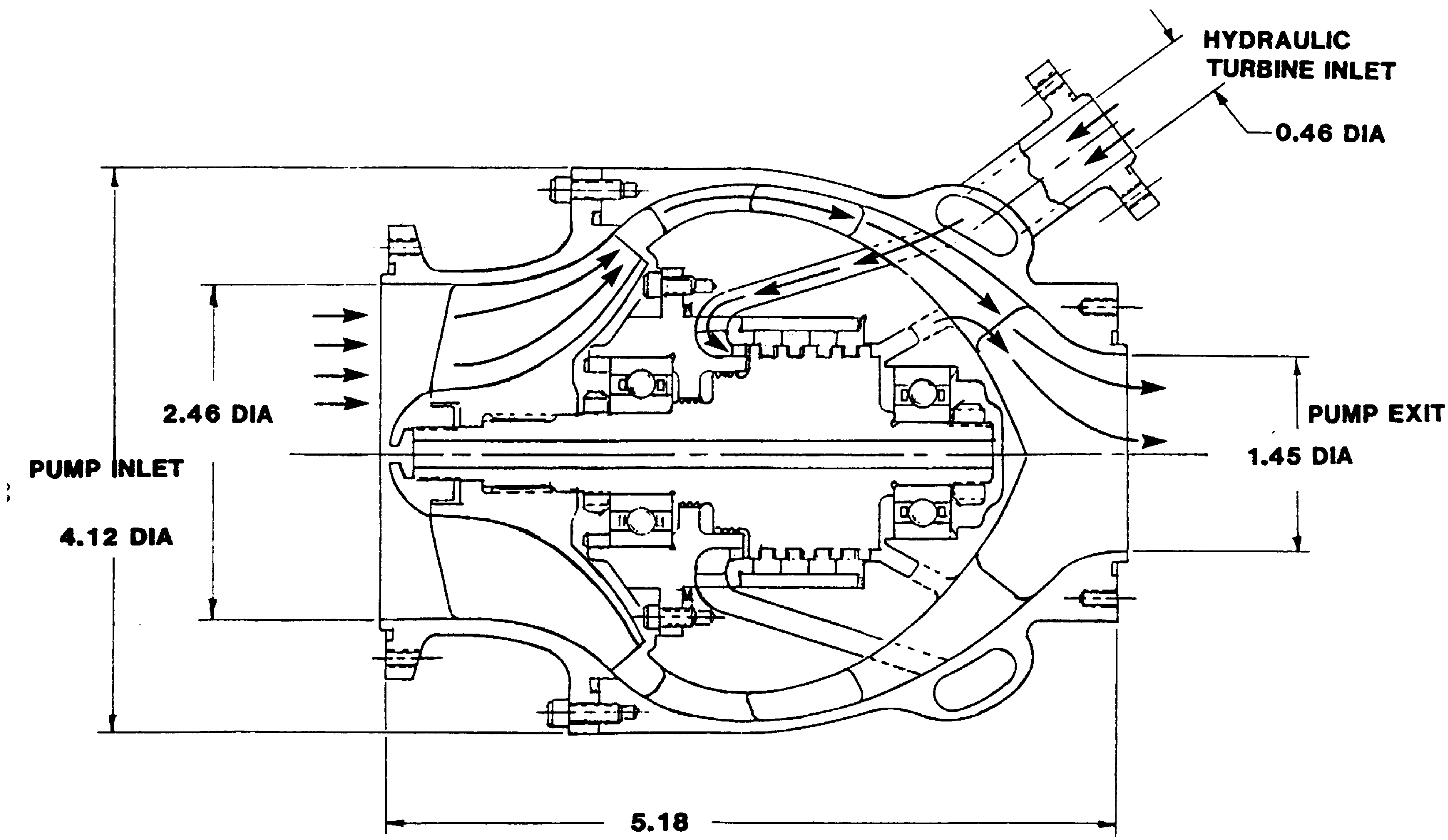


Figure 3.4.1-2 Preliminary Design OTV, 7500 Pound Thrust Engine, Boost Fuel Turbopump

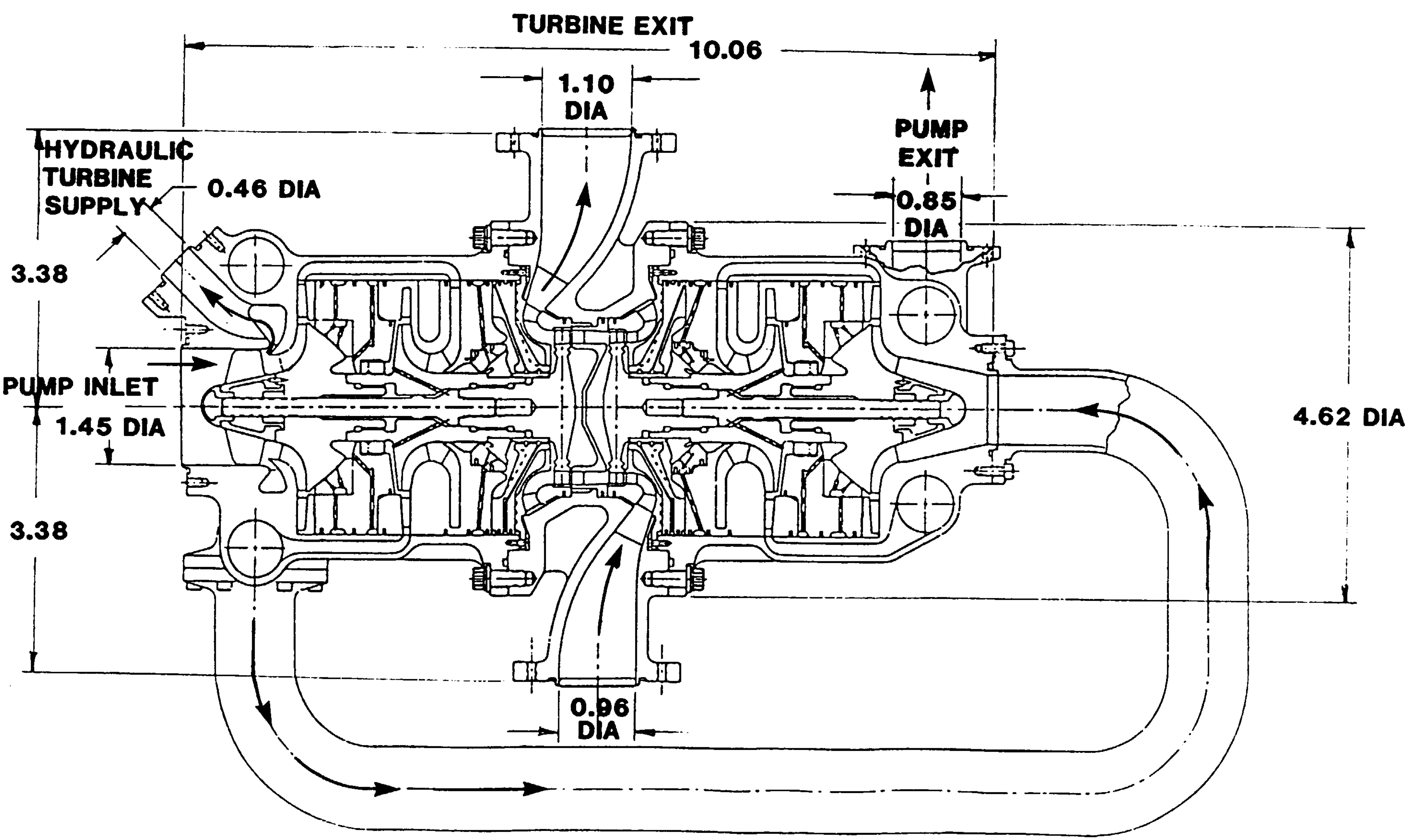


Figure 3.4.1-3 Preliminary Design OTV, 7500 Pound Thrust Engine, High Pressure Fuel Turbopump

TABLE 3.4.1-1

TURBOPUMP DESIGN POINT CONDITIONS

		Oxidizer	Fuel
Engine inlet temperature	°R	162.7	37.8
Engine minimum net positive suction head	Ft	2	15
Pump flow rate	lbs/sec	16.2	2.3
Pump discharge pressure	psia	5247	5589
Turbine inlet temperature	°R	860	1005
Turbine inlet pressure	psia	4937	4889
Turbine flow rate	lbs/sec	15.65	2.2

TABLE 3.4.1-2

PRELIMINARY DESIGN SPECIFICATION SHEET
EXTREME OPERATING PERFORMANCE

Description - Hydrogen Turbopump
OTV 7.5K Pound Thrust Engine
Throat Area 1.767 in.²
8625 Pounds Thrust 7 M.R. (1)
2300 psi Chamber (2)

Pumps	Dimensions	Boost	Inducer	Main
Propellant (1)		LH_2	LH_2	LH_2
Propellant Temperature	°R	37.8 (1)	38	40.5
Propellant Density	lb/ft³	4.38	4.36	4.3
Shaft Speed	rpm	20,000	180,000	180,000
Total Discharge Pressure	psia	39.8	221	5589 (2)
Total Suction Pressure	psia	18.5	39.8	212
Total Pressure Rise	psi	21.3	182	5377
Total Head Rise (Cavitating)	ft	700	7000	160,310
Weight Flow	lb/sec	2.3	2.78	2.3 (2)
Capacity	gpm	238	309	238
Specific Speed (Stage)	rpm x gpm / ft	2270	4400	1020
Efficiency	%	70	70	60
Fluid Horsepower	h.p.	2.92	30	670
Shaft Horsepower	h.p.	4.20	43	1117
Net Positive Suction Head	ft	15 (1)	515	6000
Suction Specific Speed	rpm x gpm / ft	30,400	20,700	5000
Thermodynamic Suppression Head	ft	7	260	
Number of Stages per Spool		1	1	2
Number of Spools		1		2
Diameter of Impeller	in.	2.46	1.29	2.12
Blade Height Exit	in.			

Reference (1): OTV Directive & Work Plan, P², Table I (2): Extreme Operating Conditions

Turbine	Dimensions	Boost	Main
Bypass	%	--	4.0 (2)
Gas		LH_2	GH_2
Shaft Power	h.p.	4.2	
Gas Weight Flow	lb/sec	0.71	2.208
Gas Inlet Total Temperature	°R		1005 (2)
Pressure Ratio		--	1.82
Static Back Pressure	psia	40	2680 (2)
Shaft Speed	rpm	20,000	180,000
Efficiency	%	50	63 & 60
Gas Inlet Total Pressure	psia	200	4869 (2)
Nozzle Area (Effective)	in.²	0.31	
Specific Heat	Btu/lb°R		3.5
Specific Heat Ratio			1.4
Gas Constant	ft/°R		772.5
Diameter	in.	1.5	1.78
Blade Height	in.	0.08	0.08

3.4.1.2 Hydrogen Turbopump Performance Models

Hydrogen and oxygen turbopump components are designed to Aerojet TechSystems Company proprietary criteria except for centrifugal pumps. Centrifugal pumps are designed to "HEAD". This analytical model has been developed at Aerojet TechSystems Company over a two year period for the purpose of making detailed performance assessments of small low specific speed pumps. The model has been validated by correlation with extensive test results acquired from company-sponsored programs.

The model is used to analyze the overall performance of the pump by examining the detailed loss mechanisms. The losses are divided into four basic categories: slip (deviation of the fluid from the discharge blade angle), hydraulic or primary flow friction losses, volumetric or leakage losses, and disc friction losses. These losses and their functional dependencies are summarized in Table 3.4.1-3. The detailed geometry of a particular pump design determines its loss distribution and resulting efficiency and input power requirements. Bearing and seal losses are not included in this model and are, therefore, not represented by pump efficiency or pump input power requirements. They are evaluated separately and are accounted for in the turbine power requirements.

3.4.1.3 Hydrogen Turbopump Trade Study

The total head rise required by the 5589 psia hydrogen discharge pressure insures that more than one centrifugal pump stage will be required for the high pressure hydrogen turbopump. The question of just how many stages is answered by looking at the trade off between pump efficiency, impeller size and impeller tip speed. Typical efficiencies, impeller size and resulting impeller tip speed is displayed on Figure 3.4.1-4 plotted versus pump speed for the 2.63 lb/sec design flow rate and total pressure rise. An examination of the results indicate that three or less pump stages pushes impeller tip speed above 2000 feet/seconds, a typical upper stress limit. Increasing the number of stages has the benefit of modestly increasing the pump stage efficiency but at the cost of increasing numbers of parts and complexity. Four pump stages are selected as the optimum for the stated requirements. Splitting the baseline design of four stages divided into two spools of two stages each permits the possibility of a sub-critical speed shaft assembly design and a two stage turbine rotor configuration. The naturally high nozzle spouting velocity resulting from hydrogen turbine drive gas can use multiple staging to avoid a lower specific speed-low efficiency turbine design. The decision to employ a boost pump permits the high pressure/high speed TPA pumps to be free of suction specific limitations. The hydrogen boost pump pressure rise is

TABLE 3.4.1-3

"HEAD" - PUMP PERFORMANCE ANALYSIS MODELING SUMMARY

Analysis Consideration	Function Of	Remarks
• Pump Input Power		Does not include bearing and seal friction
• Pump Efficiency	Size, Specific Speed, Geometric Arrangement	$\frac{\text{Input Power-Losses}}{\text{Input Power}}$
• Slip	$\beta_{2\,AVG}$, Number of Blades Velocity	Difference between discharge blade angle and fluid angle, Stodola criteria
• Hydraulic Losses		
Impeller friction	Q^2, Speed, Passage Geometry	
Jet-Wake Mixing	Ideal Head	
Volute Incidence	Volute Angle, Fluid Angle, Q	
Volute Friction	Q^2, Specific Speed, Volume Geometry	
Impeller Recirculation	Flow Coefficient, Specific Speed	
Diffuser Friction	Q^2, Diffuser Geometry	
• Volumetric Efficiency	Lab. Geometry, Specific Speed Impeller Housing Gap Geometry	$\frac{\text{Pump Flow - Lab. Flow}}{\text{Pumped Flow}}$
• Mechanical Losses	Speed, Specific Speed, Size Geometric Arrangement	Disk Friction

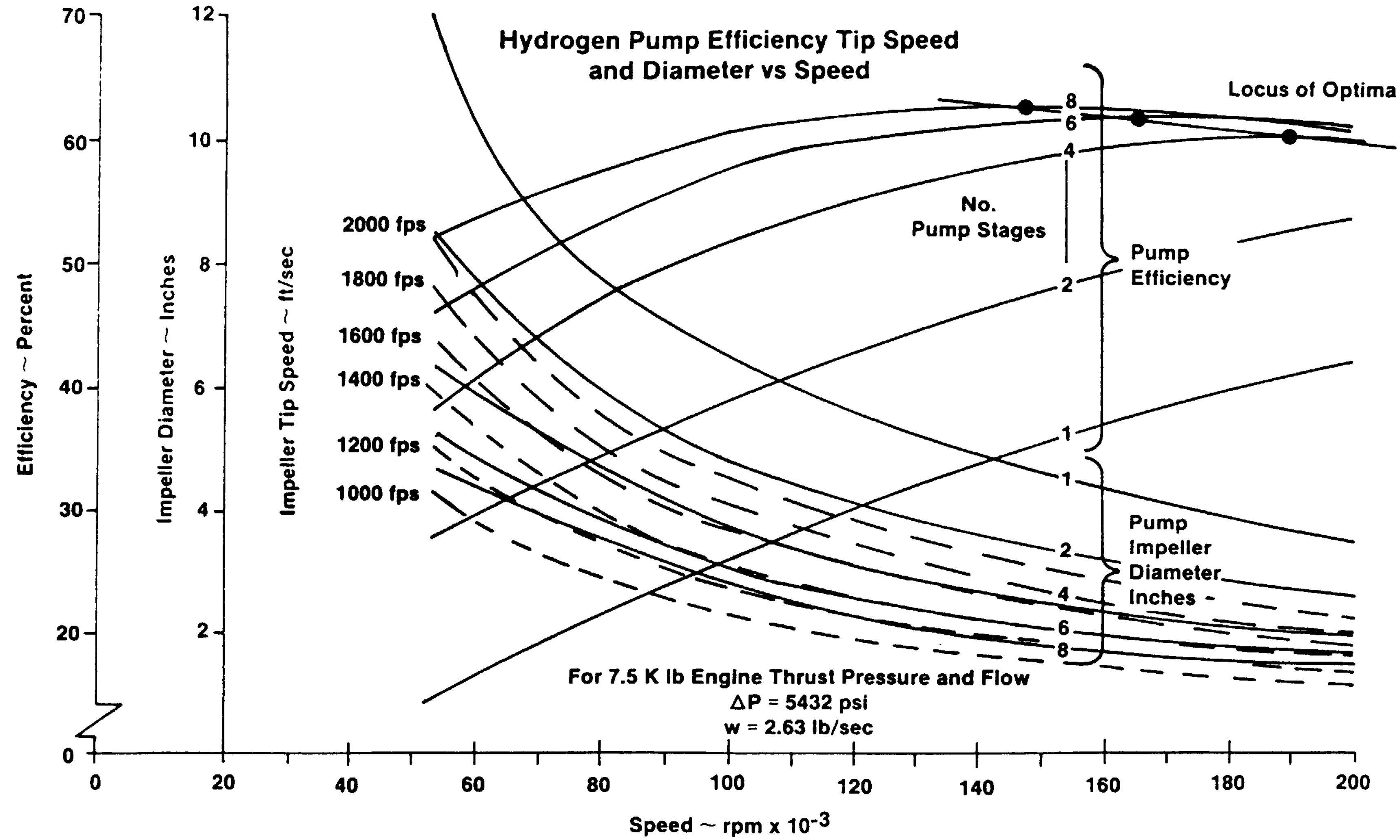


Figure 3.4.1-4 OTV Engine Preliminary Design, Optimum Hydrogen Pump Efficiency

then fixed by the high pressure turbopump inducer NPSH requirement. Figure 3.4.1-5 shows the boostpump mounted in-line ahead of the high speed TPA.

3.4.1.4 Hydrogen Turbopump Design Consideration

Design considerations for pump, bearings, seals and turbine are made on a component basis in addition to system factors. Precise component independence is not possible as each has an impact on one or more of the others. The following section comments on specific issues considered in the process of selecting the type of component and its impact on the turbopump assembly.

3.4.1.4.1 High Reliability

A primary requirement that can be varied in a machine that meets performance is reliability. The selection of the type of component and its speed (cycles) will determine that reliability for a given required design life. The updated 1987 Goals/Requirements of Table 2-1 were used for the turbopumps described in this design study. The goal of a 4 hour service life multiplied by the 5X service factor yields a 20 hour design life for the turbopump components. The assumption of bearing reliability to be 10X the 0.9997 engine reliability results in a 12,500 hour bearing design life, Table 3.4.1-4.

The component selection most significantly affected is the bearing. The small shaft diameters (17 mm) of the proposed high pressure hydrogen turbopump and the high speed (180,000 rpm) results in a ball load due to centrifugal force alone of 15 times the allowable fatigue life load limit for 12,500 hours life. Although "DN" values of 3×10^6 mm x rpm have been demonstrated in hydrogen for less than an hour and 1.5×10^6 mm x rpm for up to 10 hours, the ball bearing must be eliminated from the design of the subject high pressure turbopump on a life basis alone. Hydrostatic bearings provide the type of design that is not affected by fatigue life. Its life will be limited by wear, if significant, during the start/shutdown transient. The boost turbopump speed and loads permit ball bearings to be utilized for these lower speed machines.

3.4.1.4.2 Hydrogen Turbopump Bearings

Once the hydrostatic bearing type was selected for the high pressure hydrogen turbopump, the bearing operating characteristics were addressed. The conventional hydrostatic bearing utilizing straight journals and planor radial thrust faces is illustrated in Figure 3.4.1-5a. The thrust bearing set, however, operates with too close a running clearance for

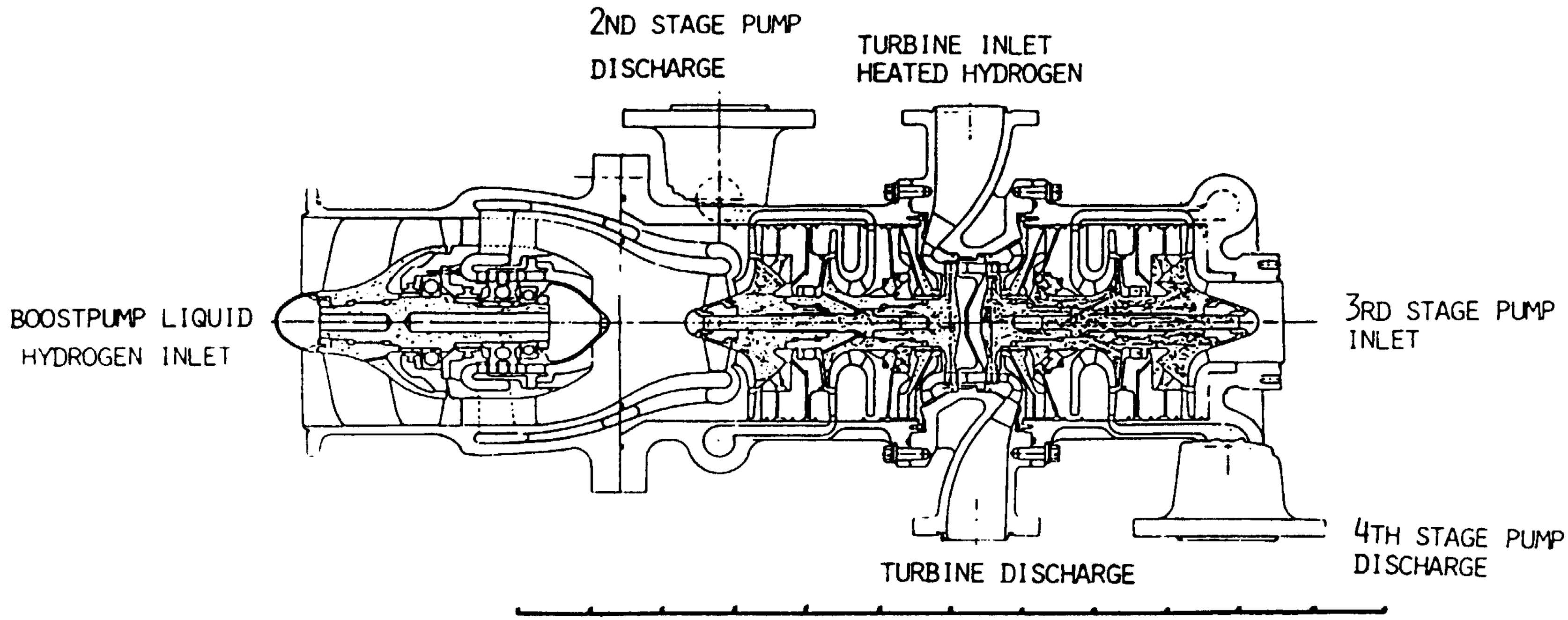


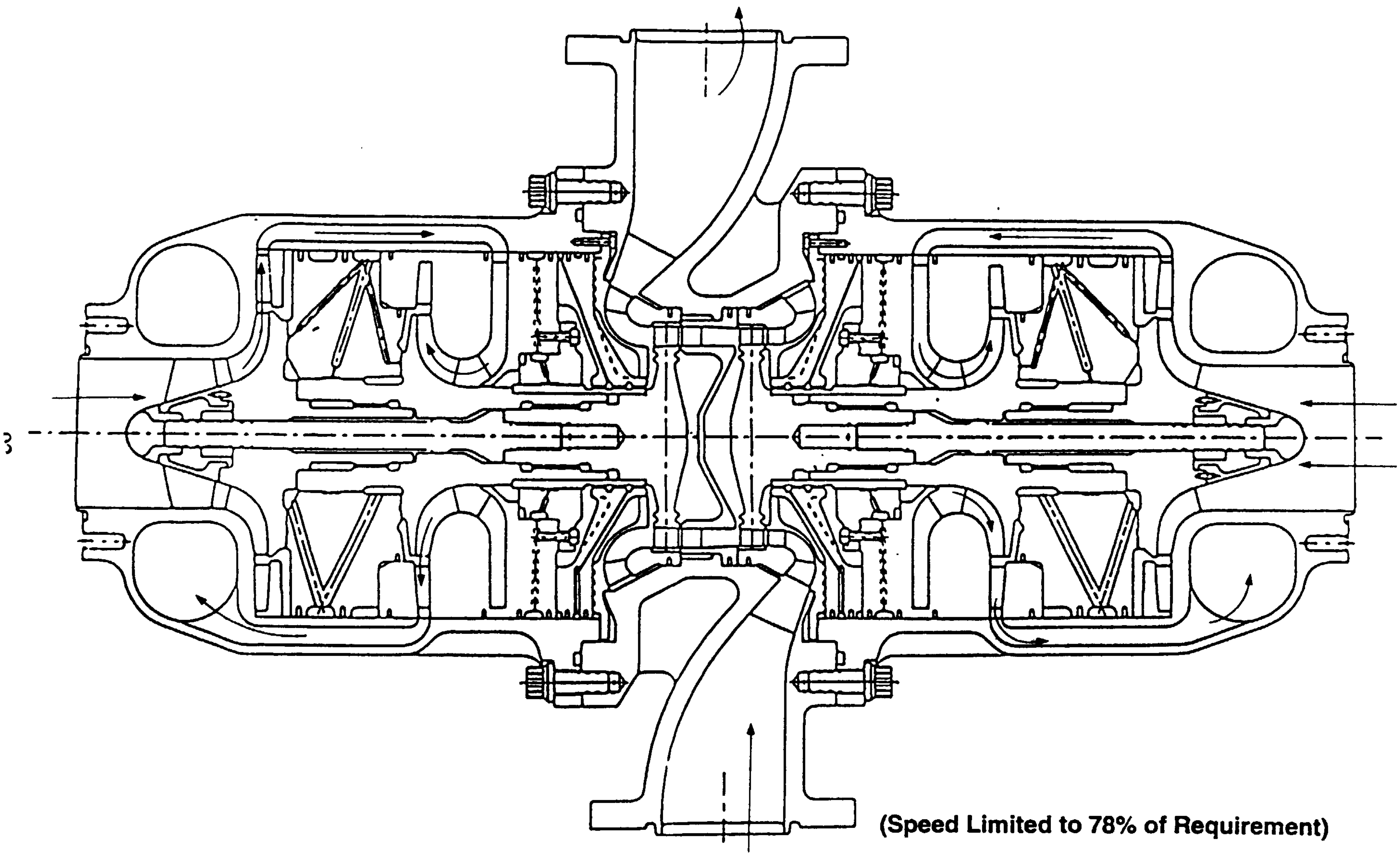
Figure 3.4.1-5 Hydrogen Bootstrap/High Pressure Turbopump Integral Assembly for 7.5K Pound Thrust

TABLE 3.4.1-4

OTV BEARING LIFE REQUIREMENT

Engine Reliability	=	0.9997
Engine Service Life	=	4 hours
Assume Bearing Reliability	≈	10 x Engine Reliability
Bearing Reliability	≈	0.99999
Bearing Design Life	=	5 x Service Life
$\therefore$Bearing B_{10} Life $\approx 5 \times 4 \times 5^4$	=	12,500 hours

Since $B_n \sim 5 \times B_n/10$



(Speed Limited to 78% of Requirement)

Figure 3.4.1-5a Hydrogen Turbopump (Conventional Hydrostatic Bearings) the shaft centerline cocking permitted by the journal bearing span. This potential interference is corrected for by making the thrust and journal bearing self-aligning via use of hydrostatically floated spherical sockets. Figure 3.4.1-6. The shaft journal sections were found to need compensation for radial centrifugal force growth to maintain the desired journal running clearance at maximum speed. This is accomplished by inserting interference fit low specific weight inserts at the journal locations. The final bearing system design incorporates axially compensated curvilinear hydrostatic bearing set, see Figure 3.4.1-3. Axial compensation was also found to be needed in addition to the radial. By hydrostatically axially loading one of the bearing surfaces, both radial and axial movement is compensated for whether it is caused by thermal growth differences or centrifugal force.

The final design configuration for the hydrogen turbopump separates the four pump stages so that two pump rotors are carried on each of two shafts. Each spool has its own turbine rotor, see Figure 3.4.1-3. The selected four stage-two spool configuration addresses the need for shaft-housing thermal length change, loss of bearing clearance due to shaft misalignment, and change in shaft radial size due to centrifugal growth. Issues to be confirmed by a feasibility program are: determine the wear coefficient and friction coefficient for the bearings, confirm the load capacity and running clearance through the start transient, and determine the critical speed margin through the start transient on up to full speed.

3.4.1.4.3 Hydrogen Turbopump Shaft/Bearing System Critical Speed

Shaft/bearing system critical speed can impact a turbopump design in numerous ways. The need to operate the unshrouded pumps at 0.001 to 0.002 inch running clearance to maintain pump efficiency and to operate over a wide if not full speed range required that shaft excursions due to passing through a critical speed be minimized if not avoided completely. Rolling contact bearings were rejected in addition due to their inability to keep deflections within the impeller/housing running clearances and their having low bearing stiffness. Figure 3.4.1-7 illustrates the relative relationship of the three typical operating zones that turbopumps are designed to operate within at full speed. Selecting zone 1 for the design range provides the capability of steady state low vibration operation over the full speed range. To obtain maximum subcritical speed, high bearing stiffness is required. The election of hydrostatic bearings provides the high bearing stiffness that permits subcritical operating speed over the speed range required.

The selection of a subcritical design also has the benefit of avoiding the potential of subsynchronous whirl and permits a lower cost low speed balance requirement.

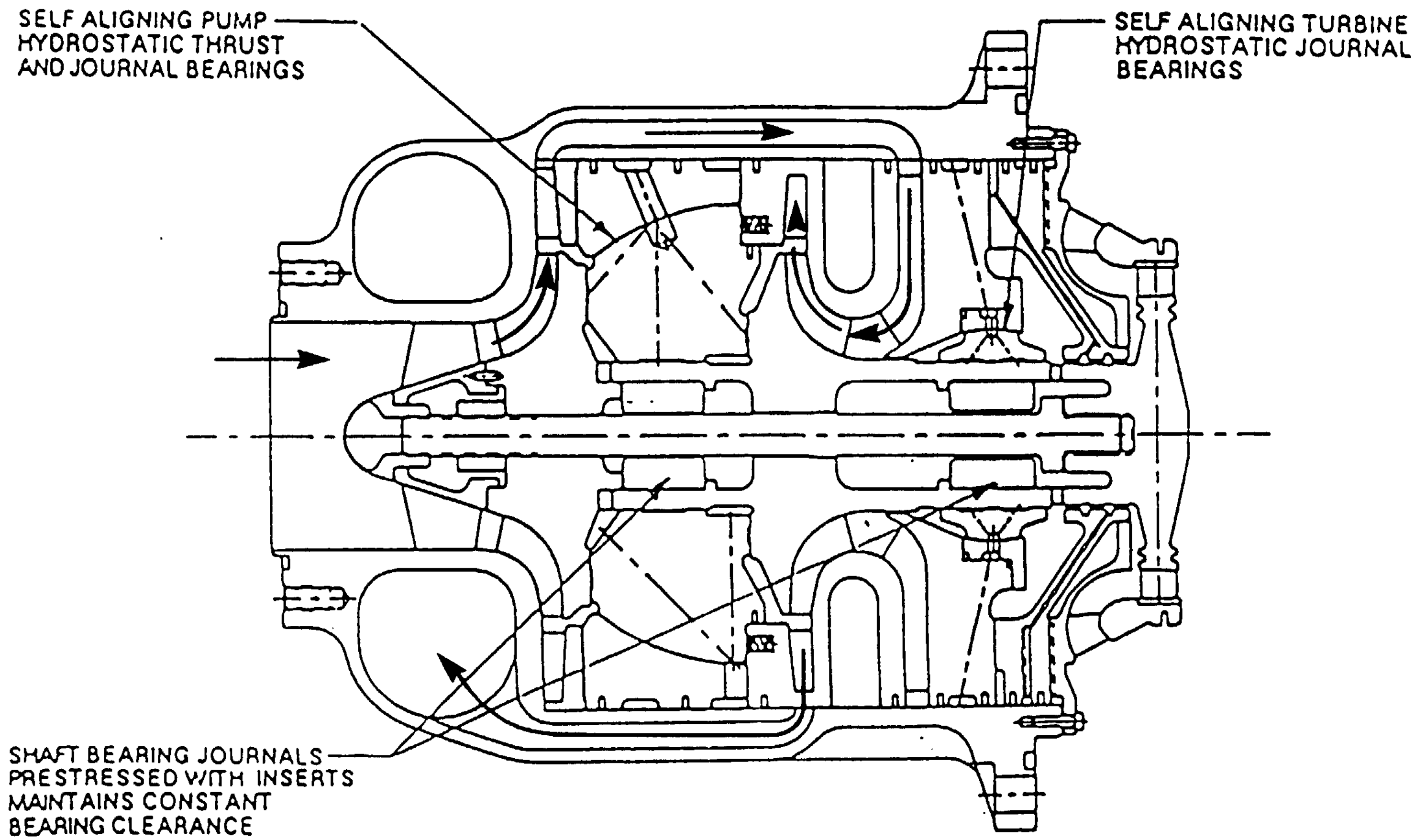


Figure 3.4.1-6 Conventional Hydrogen Turbopump Self-Aligning Bearings

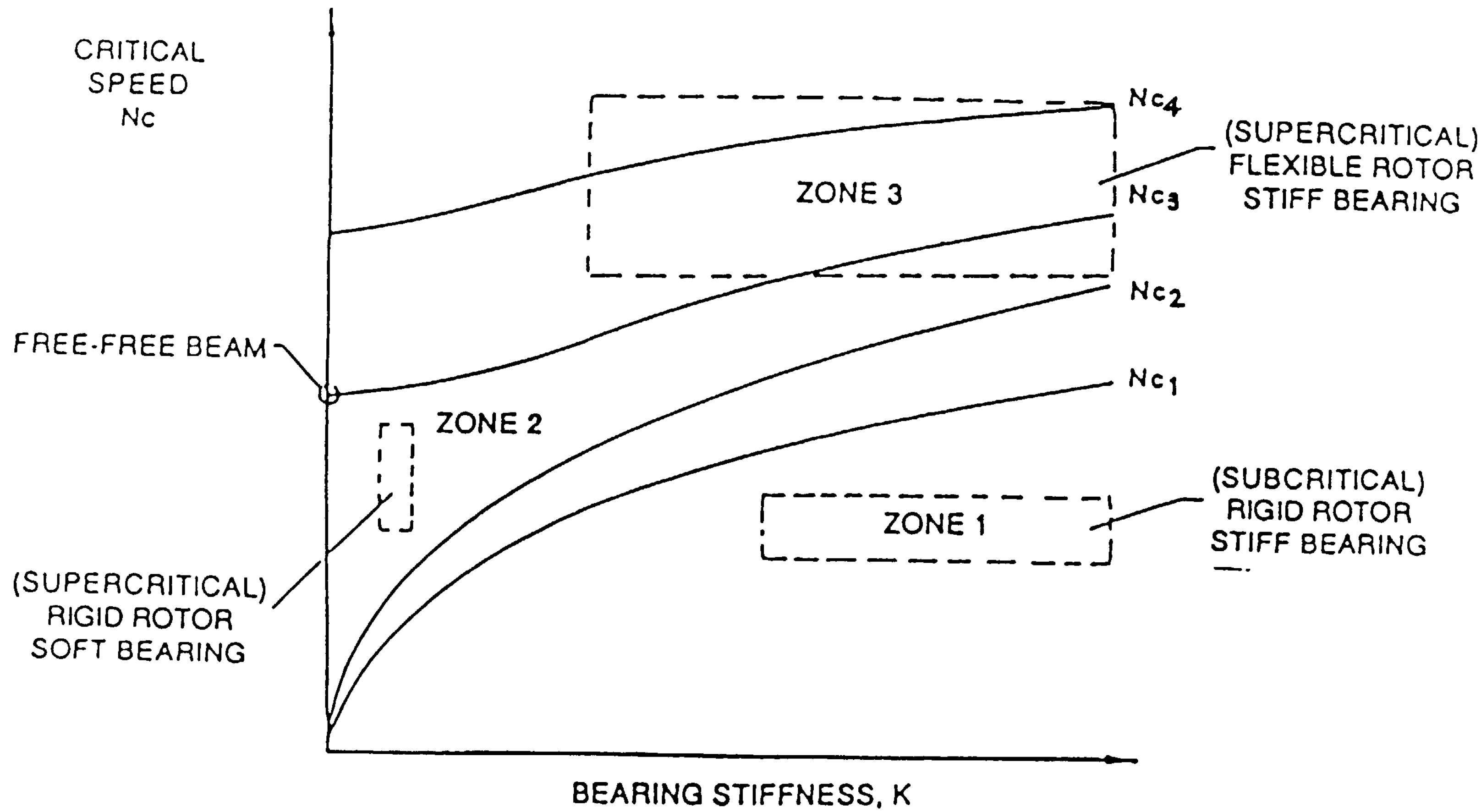


Figure 3.4.1-7 OTV Engine Preliminary Design, Critical Speed Operating Zones

3.4.1.4.4 Hydrogen Turbopump Turbines

The desire to split the hydrogen turbopump into a two spool machine is fortuitous. Whereas a single stage turbine would result in a specific speed of 21 (rpm $cfs^{1/2}/ft^{3/4}$), a more desirable specific speed of 36 each is attained as the result of using two spools (two turbine stages). Stage efficiency increases from less than 55% to greater than 60% with dual spools.

The turbines are subsonic, full admission axial flow type. Although designed as a typical two stage single shaft configuration, they will have each rotor driving two pump stages on a single shaft. Turbine rotor tip clearances of 0.001 to 0.003 inches will be practical to use with the employment of the very stiff hydrostatic bearing system. The small diameter (1.78 inch) rotors will therefore not penalize the turbine efficiency due to the typical large relative clearances of most small machines.

3.4.1.4.5 Hydrogen Turbopump Pumps

Selection of the number of pump stages was discussed in 3.4.1.3. In summary, a four stage design permits reasonable efficiency and specific speed without exceeding 1600 ft/sec tip speeds or going to impeller diameters much less than two inches.

Both shrouded and unshrouded impellers are considered. The selection of high spring rate hydrostatic bearings permits the application of unshrouded impellers without a large efficiency loss. In addition, fabrication is made easier by avoiding the attachment difficulty of a shroud.

Vaned diffusers are used as they are required as axial structural bridging between housing inserts, Figure 3.4.1-3. They will have a slightly better peak efficiency but will penalize pump head rise for non-nominal specific speed operation.

The high pressure turbopump inducer will require considerable boosting of its suction pressure in order to keep to a 20,000 design suction specific speed value. The minimum engine net positive suction head, NPSH, of 15 ft is boosted to 515 ft by a separate low speed, low pressure boost pump.

The boost turbopump is designed to operate at the engine interface minimum NPSH of 15 feet. Its prime purpose is to provide the NPSH for the high pressure turbopump. The boost turbopump speed is predicted on a 30,000 suction specific speed and its head rise on the high pressure turbopump NPSH requirements, Table 3.4.1-2.

Boost turbopump power is supplied by a four stage hydraulic turbine all carried on a common shaft, Figure 3.4.1-2. Four stages were selected as a way to increase the turbine specific speed of a low power, high available head design with a light weight (small diameter) turbine.

Hydraulic power for the turbine is provided by designing the inducer of the high pressure turbopump for 30% additional flow. The inducer has sufficient head rise to meet the need of the boost turbopump turbine. The increased 30% flow is stripped off the suction line housing between the inducer and the centrifugal impeller, Figure 3.4.1-3. A half inch diameter tube is sufficient to carry the flow.

The boost turbopump may be located close coupled and inline with the high pressure turbopump or remotely as far away as the hydrogen tank. The choice of location may be considered during the vehicle layout/weight study. A system start transient model should be able to provide system characteristics that may impact the location choice.

Rolling contact ball bearings are used to carry the boost turbopump rotor assembly. Axial thrust loads are reduced by labyrinth balance disks. Bearings are cooled by the hydrogen flowing through them. Lubrication of ball/race interface is by the solid film teflon wiped from the Armalon cage.

3.4.1.4.6 Hydrogen Turbopump Material Selections

The preliminary material selections are tabulated by component in Table 3.4.1-5 for the hydrogen boost turbopump. The selection logic is driven in large part by use of bearing sets with BG42 races and 440C balls, a proven combination for hydrogen submerged bearings. The bearing cage is reinforced "teflon" ("Armalon") a proven boundary lubricant supplier. The shaft and bearing housing selection is a titanium alloy used to obtain a lightweight material of similar coefficient of expansion to BG42/440C. Hydrogen embrittlement should not be a problem as exposure to hydrogen is mostly at -400°F temperatures. Aluminum alloys will be used for the pump impeller and housing castings. The low inducer/impeller tip speed of 200 feet/second and modest housing pressure permit the low stress capability of aluminum to be utilized.

The high pressure hydrogen turbopump material selections are noted on Table 3.4.1-6. They consist of titanium alloys for the cold hydrogen areas and A-286 for the hot turbine parts. Titanium high strength/weight ratio property at liquid hydrogen temperatures are

TABLE 3.4.1-5

OTV BOOST TURBOPUMP
PRELIMINARY DESIGN MATERIAL SELECTION LIST BY
COMPONENT FOR HYDROGEN SERVICE

Component	Material
Housings	Aluminum
Inducer - Impeller	Aluminum
Turbine Rotor - Shaft	Titanium
Turbine Stator - Shroud	Titanium
Rolling Contact Bearing	BG42 Race/440C Ball
Bearing Separator	Armalon
Bearing Housing	Titanium

TABLE 3.4.1-6

OTV HIGH PRESSURE TURBOPUMP
PRELIMINARY DESIGN MATERIAL SELECTION LIST BY
COMPONENT FOR HYDROGEN SERVICE

Component	Material
Housings	Titanium
Inducer - Impeller	Titanium
Hydrostatic Bearing	Titanium
Interstage Seal	Bronze
Turbine Housing/Nozzle	A-286
Turbine Rotor - Shaft	A-286
Shaft Tie Bolt	A-286

required to meet the 1700 feet/second pump rotor tip speed stress requirements. To keep bearing and bearing housing linear expansion coefficientts the same, these parts will also be titanium alloy. The high temperature/high stress turbine parts are to be A-286, a good high strength-high temperature alloy that is compatible with hydrogen. The shaft tie bolt will utilize A-286 for its high strength at liquid hydrogen temperatures while maintaining its ductility. The higher linear expansion coefficient of A-286 relative to titanium alloy will provide additional bolt loading when cooled to liquid hydrogen temperature.

Hydrostatic bearing interface material selection will require backup materials to safeguard against a bearing wear problem.

Alternate material pairs are therefore noted in Table 3.4.1-7 for the hydrogen bearing interface surfaces. They have characteristics and/or empirical wear data that suggest they may have good wear resistance when rubbed in a liquid hydrogen environment.

3.4.1.4.7 Hydrogen Turbopump Design Conclusions

The hydrogen turbopump pumping system design philosophy for the proposed engine preliminary design is summarized as follows:

1) The 1987 updated goals/requirements led to the decision to design two turbopumps working in series in order to reduce the weight of the turbomachinery. The first pump will boost the pressure sufficiently to prevent the high pressure turbopump from being speed limited by suction specific speed. The combined weight of the boost and high pressure turbopumps are lighter than a single low speed machine.

2) Subcritical shaft operating speed over the full pressure-flow requirements is necessary to achieve the 10 to 1 thrust throttling ratio. A subcritical design will permit constant speed operation at any pressure-flow condition that the engine may be called to operate at without producing shaft motions that result in rotor assembly rubs.

3) The high reliability of 0.9997 at 90% confidence level requires the use of fluid film bearings to avoid the limitation of rolling contact bearing fatigue life and cage wear.

4) The bearing loads and stiffness requirements at design specific speed and off design specific speed require the use of hydrostatic bearings to avoid impeller and turbine rotor rubs.

TABLE 3.4.1-7

FUEL TURBOPUMP
ALTERNATE BEARING MATERIAL OPTIONS

Rotating Member	Bearing
Titanium	Silver Plate/Titanium Gold Plate/Titanium
Molydisulfide (Microseal 200-1)*	Molydisulfide (Microseal 200-1) Titanium Tiodize/Titanium
Electrolize Chrome/Titanium	Molydisulfide/Silver Plate/ Titanium Titanium
Tiodize/Titanium	440-C (BG42) Leaded Bronze Carbon/Graphite Silicon Carbide Stellite 6B

* Molydisulfide (Microseal 200-1) is a solid film lubricant which is applied by a spraying process and subsequently cured to assure adhesion.

5) The close bearing operating clearances of 0.00003 to 0.00005 inch will require the application of self aligning and axial adjusting bearing systems to accommodate tolerance and thermal growth differentials.

3.4.2 Oxygen Turbopump Assemblies

The oxygen pressurizing system consists of a low pressure boost turbopump followed by a high pressure turbopump, Figure 3.4.1-1. The boost turbopump, Figure 3.4.2-1, consists of a single stage pump driven by a four stage hydraulic turbine on a common shaft. The rotor assembly is carried by a set of ball bearings which are cooled by the pumped liquid oxygen. Labyrinth seals are used to maintain pressure differential between turbine-bearing-pump cavities. Limited leakage is permissible as the liquid oxygen flows in all components. The boost turbopump is estimated to weight 6 pounds.

The high pressure oxygen turbopump consists of a two stage centrifugal pump with first stage inducer driven by a single stage turbine. The single shaft rotor assembly is carried on axially compensated hydrostatic spherical bearings flowing liquid oxygen. Bearing flow is supplied by the high pressure pump stage, Figure 3.4.2-2. Estimated weight is 9 pounds.

3.4.2.1 Oxygen Turbopump Design Criteria

Oxygen turbopump design operating conditions are also dictated by the OTV engine interface requirements, Table 2-1 and calculated engine parameters. The resulting engine flow rates and pressures, tabulated in Table 3.4.2-1, are the basis for the oxygen turbopump design extreme operating conditions. The basic through flow rates and pump discharge pressures result from engine requirements (see power balance, Section 3.1). Suction pressure and temperature are based on engine interface requirements. Turbine flow rate and pressure ratio result from the available turbine inlet pressure and temperature dictated by the thrust chamber oxygen cooling system heating and pressure drop characteristics.

The oxygen turbomachinery speed is also controlled by the specified engine interface requirements. The specified minimum NPSH of 2 feet requires the first pump to be limited to 6200 rpm. A dual turbopump system is therefore baselined for the oxygen pumping system for the same reasons as the hydrogen system. The design extreme operating performance is summarized in Table 3.4.2-1.

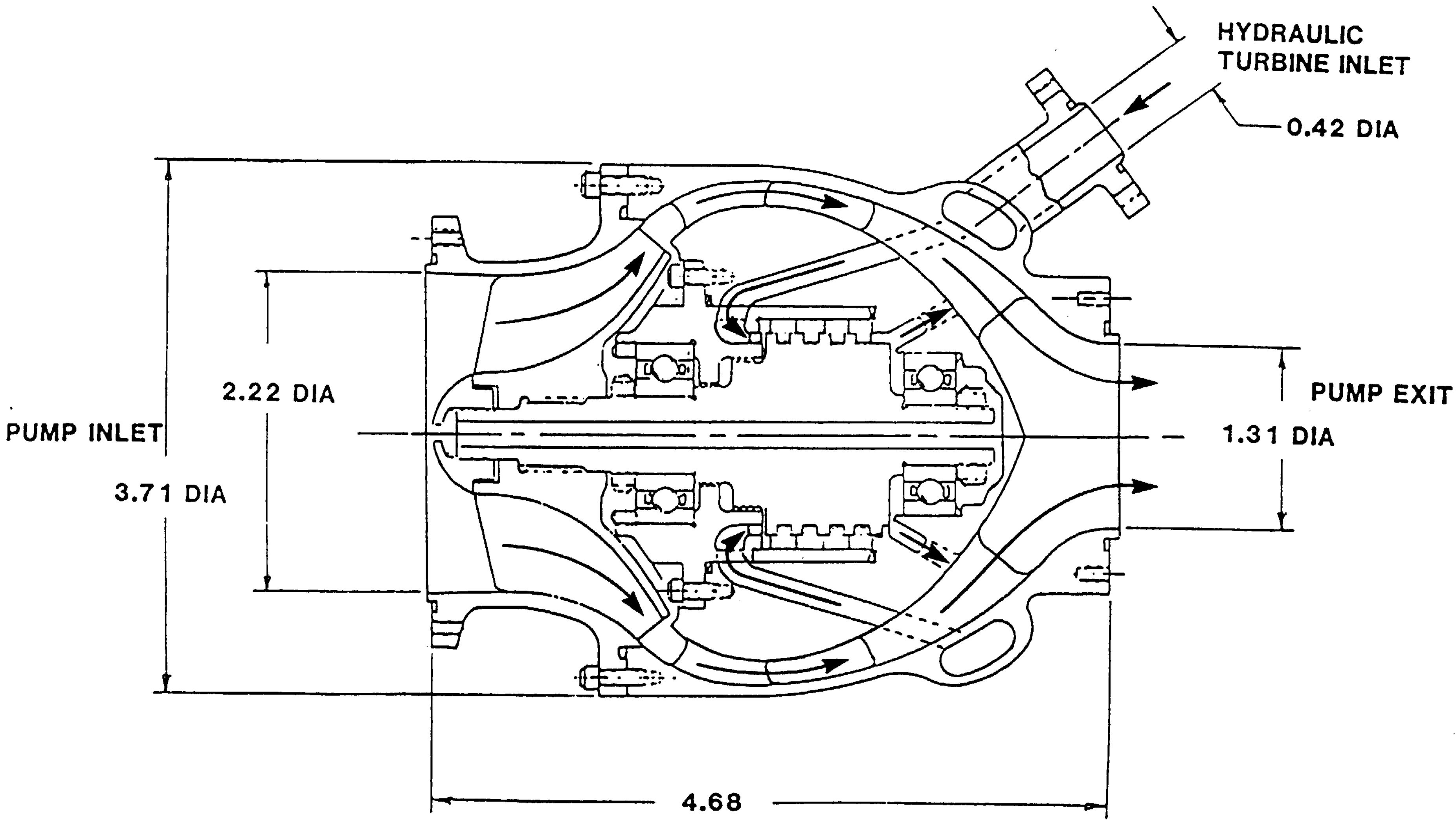


Figure 3.4.2-1 Preliminary Design OTV 7500 Pound Thrust Engine Booster Oxidizer Turbopump

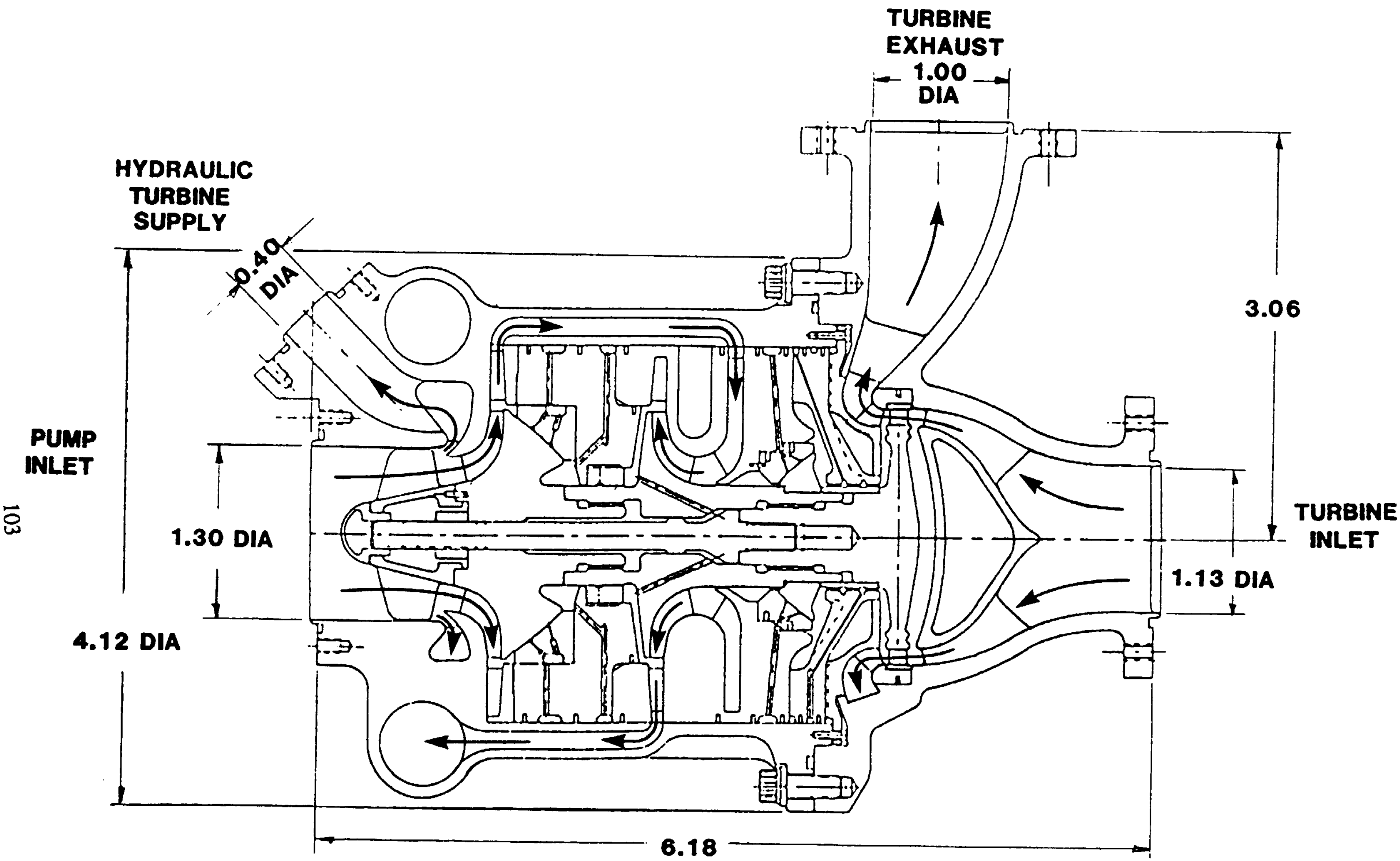


Figure 3.4.2-2 Preliminary Design OTV, 7500 Pound Thrust Engine, High Pressure Oxidizer Turbopump

TABLE 3.4.2-1

PRELIMINARY DESIGN SPECIFICATION SHEET
EXTREME OPERATING PERFORMANCE

Description - Oxygen Turbopump
OTV 7.5K Pound Thrust Engine
Throat Area 1.767 in.²
8625 Pounds Thrust 7 M.R. (1)
2300 psi Chamber (2)
Revised 2/18/88

Pumps	Dimensions	Boost	Inducer	Main
Propellant (1)		LO_2	LO_2	LO_2
Propellant Temperature	°R	162.7 (1)	163	165
Propellant Density	lb/ft³	71.17	71	71
Shaft Speed	rpm	6200	70,000	70,000
Total Discharge Pressure	psia	44.7	494.7	5247 (2)
Total Suction Pressure	psia	15.7	44.7	494.7
Total Pressure Rise	psi	22.9	450	4742
Total Head Rise (Cavitating)	ft	58	900	8742
Weight Flow	lb/sec	16.2	19.4	16.2 (2)
Capacity	gpm ℓ)	102.1	122	102.1
Specific Speed (Range)	$\frac{\text{rpm x gpm}}{\text{ft}}$	3000	4700	1200
Efficiency	%	70	70	65
Fluid Horsepower	h.p.	1.47	31.74	287
Shaft Horsepower	h.p.	2.1	45.35	441
Net Positive Suction Head	ft	2 (1)	58	955
Suction Specific Speed	$\frac{\text{rpm x gpm}}{\text{ft}}$	32,000	30,000	4,000
Thermodynamic Suppression Head	ft	0.45	16	16
Number of Stages per Spool		1	1	2
Number of Spools		1	1	1
Diameter of Impeller	in.	2.74	1.31	1.95
Blade Height Exit	in.			

Reference (1): OTV Directive & Work Plan, P2, Table I (2): Extreme Operating Conditions

Bypass	%	--	3.4 (2)
Gas		LO_2	GO_2
Shaft Power	h.p.	2.1	486
Gas Weight Flow	lb/sec	3.2	15.65
Gas Inlet Total Temperature	°R		860 (2)
Pressure Ratio			1.86
Static Back Pressure	psia	45	2645 (2)
Shaft Speed	rpm	6200	70,000
Efficiency	%	50	63
Gas Inlet Total Pressure	psia		4937 (2)
Nozzle Area (Effective)	in.²		0.626
Specific Heat	Btu/lb°R		0.248
Specific Heat Ratio			1.4
Gas Constant	ft/°R		48.28
Diameter	in.	1.7	1.82
Blade Height	in.	0.09	0.127

3.4.2.2 Oxygen Turbopump Performance Models

The performance models used for the oxygen turbopump components are the same as used for the hydrogen turbopumps. See paragraph 3.4.1.2 for comments.

3.4.2.3 Oxygen Turbopump Trade Study

The efficiency-diameter-tip speed trade study yielded the optimum efficiency at 110,000 rpm for the high pressure turbopump employing two pump stages. Pump impeller diameters at this speed are 1.25 inches over all. Although this small size presents no performance problem in itself (rotor Reynolds Number >30 x 10^6) the bearing running clearances are getting very small. Manufacturing cost due to small tolerances does become a factor when making turbomachinery with 1 inch rotors and smaller. A design speed of 70,000 rpm is therefore selected for the proposed design. It yields an impeller just under 2 inch diameter that has only a 4 percentage point lower efficiency than the 1.25 inch diameter design. The 70,000 rpm design speed is therefore believed to be a good compromise between performance and practical manufacturing cost and fabrication difficulty.

The possibility of some commonality between the fuel and oxidizer turbopump assemblies is possible. One spool of two pump stages and a single stage turbine are very close in physical size and stage specific speed to one of the two spools of the hydrogen turbopump assembly. Although material selection, fits/clearances effected by temperature and turbine nozzle area can be different between oxygen and hydrogen turbopumps, most parts can be geometrically the same and, indeed, the same size. This should be investigated on the next design point analysis cycle for its potential reduction in design and fabricating costs.

3.4.2.4 Oxygen Turbopump Design Considerations

The following section comments on the design considerations that are different from the hydrogen turbomachinery. Otherwise the oxygen turbomachinery is designed to the same criteria as the hydrogen turbopump.

3.4.2.4.1 Reliability

The oxygen turbopump shaft requires a 20 mm bore ball bearing. The centrifugal ball load force at 70,000 rpm is about 12 times the rated oil lubricated fatigue life load. Oxygen bearing experience has demonstrated 1 million DN (mm x rpm) for 15 hours, about 70% of the DN value required. Bearing reliability requirements of 0.99999 at 90% confidence level for

4 hours service life (20 hour design life) also require that the oxygen high pressure turbopump employ hydrostatic bearings to avoid fatigue and wear life limitations of rolling contact bearings. There is the potential of some wear taking place as a result of the start/shutdown transient operation. The degree of this wear and methods of minimizing it will need to be determined during a development program.

The boost turbopump operating speeds will permit rolling contact bearing to be utilized.

3.4.2.4.2 Oxygen Turbopump Bearings

The primary design selected for this study is the axially compensated curvilinear bearing as shown in Figure 3.4.2-2. The current hydrostatically self aligning spherical socket design of the 3.5K pound thrust size orbit transfer vehicle turbopump, Figure 3.4.2-4, is applicable to the proposed 7.5K pound thrust engine oxygen turbopump, but the axially compensated curvilinear hydrostatic type proposed for the high pressure hydrogen turbopump will also meet the requirements. Although its axial compensation feature may not be required for an oxygen turbopump, it may prove to be less costly to manufacture. The self aligning spherical socket design demonstrated in the 3.5K pound thrust OTV oxygen turbopump can be considered as a backup design.

3.4.2.4.3 Oxygen Turbopump Shaft/Bearing System Critical Speed

The discussion of the hydrogen high pressure turbopump also applies to the oxygen turbopump. See Section 3.4.1.4.3. The oxygen turbopump will utilize only one spool instead of the two required for the higher head hydrogen pumps.

3.4.2.4.4 Oxygen Turbopump Turbine

The oxygen high pressure turbopump turbine is a single stage full flow axial flow type. As only a single two pump stage spool is required one turbine stage is sufficient. Running rotor tip clearances can be kept to 0.002-0.003 inch which will minimize the losses usually occurring with small size machines. Operating performance for the extreme operating conditions are tabulated on Table 3.4.2-1. The Specific Speed is 40 and Specific Diameter 1.6 yielding an estimated efficiency of 63%.

Flow bypassed around the turbine to allow for speed control tolerance is 3.4%. Turbine gas flow velocity reaches a maximum of Mach number 0.96 at the inlet nozzle.

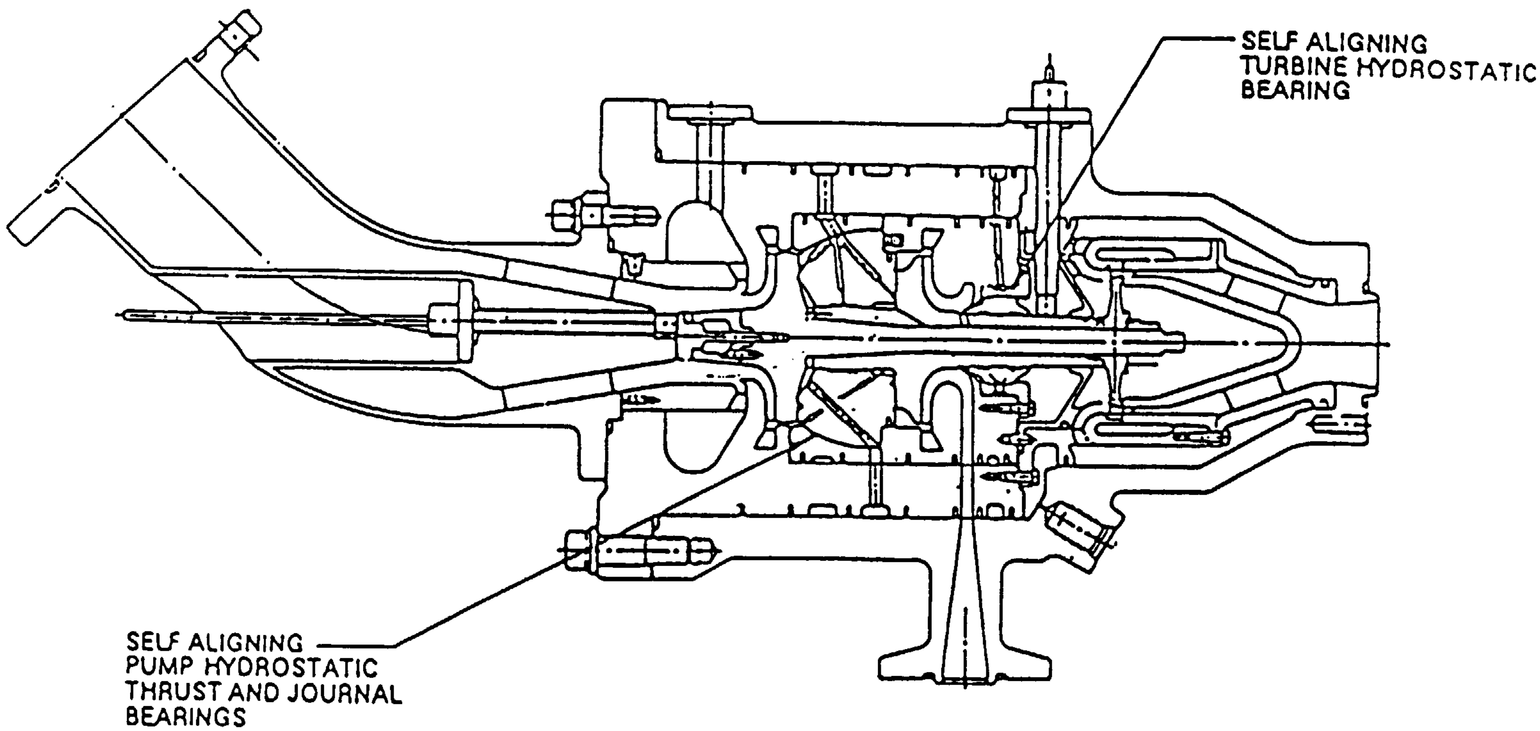


Figure 3.4.2-4 OTV Oxygen Conventional Self-Aligning Hydrostatic Bearings (3.5K Size)

3.4.2.4.5 Oxygen Turbopump Pumps

The oxygen high pressure turbopump uses two pump stages on a single shaft, Figure 3.4.2-2. The impellers are unshrouded backward centrifugals placed back to back to balance axial thrust. The first stage pump utilizes an axial flow inducer to bring the suction pressure up for the centrifugal stage and also provide hydraulic flow to the low pressure boost pump turbine. The high pressure turbopump is able to run at 70,000 rpm with the lower speed, 6200 rpm, boost pump providing a modest 23 psi pressure boost. As the boost turbopump gets its hydraulic power from the high pressure turbopump it will keep its speed ratio relative to the high pressure turbopump and provide increasing high pressure turbopump suction pressure with increasing speed. Modest thermodynamic suppression head was added to the NPSH for calculating Suction Specific Speed numbers.

The boost turbopump, Figure 3.4.2-1 is a single stage centrifugal type with its inducer integrated into a single hub rotor.

3.4.2.4.6 Oxygen Turbopump Material Selections

Materials selected for the boost and high pressure turbopumps are done primarily on a materials oxygen compatibility basis. Monel 400 and K500 are utilized for the impellers and housings, Tables 3.4.2-2 and 3.4.2-3. Nickel 200 is used in the lower strength requirement parts that may sustain a rub. Exceptions are the ball bearings of the boost turbopump. They are the conventional BG42/440C races and balls that employ Armalon glass reinforced cages. Parts not contacting oxygen will use A-286 where high strength is required. Silver will be used where parts are expected to rub: pump impeller blade tips and turbine blade tips.

3.4.2.4.7 Oxygen Turbopump Design Conclusions

The oxygen turbopump pumping system design philosophy for the proposed engine preliminary design is summarized as follows:

1) The 1987 updated goals/requirements for a 7500 lbf thrust oxygen/hydrogen engine led to the design of two turbopumps working in series in order to reduce the total weight of the turbomachinery. The first pump will boost the pressure sufficiently to avoid the high pressure turbopump from being speed limited by Suction Specific Speed. The combined weight of the boost turbopump and the high pressure turbopump are lighter than a single low speed machine.

TABLE 3.4.2-2

OTV BOOST TURBOPUMP
PRELIMINARY DESIGN MATERIAL SELECTION LIST BY
COMPONENT FOR OXYGEN SERVICE

Component	Material
Housings	Monel 400
Inducer - Impeller	Monel K-500
Rolling Contact Bearing	BG42 Race/440C Ball
Bearing Separator	Armalon
Bearing Housing	Nickel 200
Turbine Stator Shroud	Nickel 200*
Turbine Rotor - Shaft	Monel K-500

*Silver insert or plating at potential contact surfaces

TABLE 3.4.2-3

OTV HIGH PRESSURE TURBOPUMP
PRELIMINARY DESIGN MATERIAL SELECTION LIST BY
COMPONENT FOR OXYGEN SERVICE

Component	Material
Housings	Monel 400
Inducer - Impeller	Monel K-500
Hydrostatic Bearings	Monel K-500*
Interstage Seal	Nickel 200*
Turbine Shaft Seal	Nickel 200*
Turbine Tip Seal	Nickel 200*
Turbine Housing/Nozzle	Monel 400
Turbine Rotor - Shaft	Monel K-500
Shaft Tie Bolt	Monel K-500 Coated

*Silver insert or plating at potential contact surfaces

2) Subcritical shaft operating speed over the full pressure-flow requirements is necessary to achieve the 10 to 1 thrust throttling ratio. A subcritical design will permit constant speed operation at any stable pressure/flow condition that the engine may be called upon operate.

3) The high reliability of 0.9997 at 90% confidence level forced selection of a hydrostatic bearing system as no rolling contact bearing system could meet the life requirements.

3.4.3 Valves

This engine design requires eleven valves. Two more, the back pressure valves, might also be needed if a LO_2/GH_2 Heat Exchanger (HEX) design can not effectively handle the oxygen phase change in a predictable manner. Table 3.4.3-1 lists the valves, their functions, and some information on environments and design. Single page specification sheets with an envelope drawing are included in Appendix A.3 to present the information normally used for an initial vendor survey for procurement or development at the start of a detail design program with component verification testing.

The large number of valves is determined by the engine requirements for throttling and tank head start. There are no redundant valves. Manrating requirements for multiple valves are considered to be satisfied by additional propellant flow control valves upstream of the main propellant shutoff valves and by a health monitoring system that will shutdown an engine should a defective valve lead to a condition where the engine could be damaged. Proper sequencing must be done by the controller using pre-established timing and sensor data. As this is a closed loop system all electrically operated valves are assumed to have a circuit built-in that signals the valve is open, closed, or at a particular position if it is a proportioner valve. This signal is used by the controller logic, the health monitor system, and will be part of the pilot display of engine information.

3.4.3.1 Valve Trade Studies

Two trade studies were done in the valve selection process. The first was to select the actuation methods for each valve. The second was to select a valve type using the selected actuation method.

TABLE 3.4.3-1

OTV ENGINE CONTROL VALVES

Valve Name	Abbreviation	Function	Fluid	Operating Environment	Type
Oxygen Main Shutoff	OMSV	Propellant Isolation	Liquid O_2	0 to 45 psia -310 to 0°F	Powered Open/ Solenoid Latch/ Spring Loaded Closed
Hydrogen Main Shutoff	HMSV	Propellant Isolation	Liquid H_2	0 to 45 psia 40 to 400°R	Powered Open/ Solenoid Latch/ Spring Loaded Closed
Oxygen Turbine Bypass	OTBV	Thrust Control	Hot GOX	0 to 5500 psia 0 to 500°F	Servo/Pintle Fails in Place
Hyrogen Turbine Bypass	HTBV	Mixture Ratio Control	Hot GH_2	0 to 6000 psia 0 to 1000°F	Servo/Pintle Fails in Place
Hydrogen Regenerator Bypass Valve	HRBV	Temperature Control of Hydrogen Entering Chamber	Cold GH_2	0 to 5600 psia -340 to 420°F	Modulating Poppet Fails in Place
Heat Exchanger Bypass Valve	HEBV	Temperature Control of Oxygen to OX TPA Turbine	Hot GH_2	0 to 2700 psia 0 to 800°F	Modulating Poppet, Fails in Place
Hydrogen Idle Valve	HIV	Idle Operation MR Control	GH_2	80 psid, 0 to 6000 psia -360 to 540°F	Modulating Poppet, Fails Closed**
Oxygen Back Pressure	OBPV	Heat Exchanger Pressure Control	Warm GO_2	200 psid, 0 to 5000 psia -250 to 400°F	Open/Partial* Closure

TABLE 3.4.3-1 (cont.)

Valve Name	Abbreviation	Function	Fluid	Operating Environment	Type
Hyrogen Back Pressure	HBPV	Pressure Balance During Start	Warm GH_2	200 psid, 0 to 6000 psia -360 to 600°F	Open/Partial* Closure
Igniter Flow Control(2)**	OICV, HICV	Igniter Propellant Flow Control	GH_2, GO_2	0 to 6000 psia -400 to 100°F	Powered/Open Spring Loaded Closed
Oxygen Tank Pressurization	OTPV	Oxygen Tank Pressurization Control	Warm GO_2	0 to 3000 psia 0 to 400°F	Powered Open/ Spring Loaded Closed
Hydrogen Tank Pressurization	HTPV	Hydrogen Tank Pressurization Control	Warm GH_2	0 to 5000 psia 0 to 800°F	Powered Open/ Spring Loaded Closed

*Pressure operated. No electrical actuation devices. These valves are used to raise circuit pressure high enough to avoid undesired oxygen heating and consequent film boiling in the heat exchanger.

**Not shown on engine schematic.

As of September 1988.

The goals of the actuation study were to 1) minimize weight, 2) minimize power consumption, 3) maximize reliability, 4) maximize repeatability (for servo valves), and 5) assure maximum controllability. Only three actuation methods were considered: pneumatic, electrical, and hydraulic. There was a bias for an all electrical actuation system, but the valve designer was required to give a fair evaluation of competing methods. This was done by establishing a set of criteria and evaluating each method within the context of the engine use and the mission environment. This trade is outlined in Table 3.4.3-2. There are actuation specific concerns that also required consideration. They are summarized in Table 3.4.3-3. Table 3.4.3-4 gives the conclusions of the trade study. Essentially an all electrically actuated valve set is considered practical. The relatively slow actuation times needed for this engine versus the very fast times common in storable propellant rocket engines allow for a lighter weight valve design that will be weight competitive.

The second set of trades involved the selection of a specific valve type. Table 3.4.3-5 is a summary of the trades for the main propellant shutoff valves. It was concluded that both ball and blade type valves would be acceptable. The igniter and tank pressurization valves are covered in Table 3.4.3-6. A conventional poppet type valve is acceptable. The more complicated modulating valves used for engine control are evaluated in Table 3.4.3-7. A pintle type valve is considered best for the OTV application. An example of a valve of this type is shown in Figure 3.4.3-1. The design shown has two separate motors for complete electrical redundancy. The structure between the motors is a device for measuring valve position. Note that it has an electrical connection. Aerojet considers these valves to be excellent candidates for a development program to reduce size, weight, and mechanical complexity.

One of the valves used in the start sequence and for low thrust operation is the hydrogen idle valve. The trade on it is presented in Table 3.4.3-8. The high pressure sealing demands for this valve led to the selection of a ball valve that would be spring loaded closed if there were an electrical failure. The concern here was to assure that a normally operating engine would not be shutdown by a power failure to this valve. It must be operational, however, during start as it provides mixture ratio control until the turbine bypass valve becomes effective. A version of this valve is shown in Figure 3.4.3-2.

Two valves can possibly be eliminated from the engine. They are the back-pressure valves used to raise system internal pressure above two-phase conditions for the oxygen conversion from low temperature liquid to high temperature gas. They function as high pressure drop orifices that move to a low pressure drop "open" position automatically as system pressure

TABLE 3.4.3-2

VALVE ACTUATION TRADE STUDY CRITERIA

Criterion

1. Complexity

   A. Pneumatic

      1) Relatively simple; can use high pressure stored He, or H_2 gas
      2) Requires vent system for H_2 gas
      3) Periodic servicing necessary with service parts/lines/gauges
      4) Monitoring required of stored gas pressure

   B. Electrical

      1) Requires high amperage lines and controls for motor
      2) Simple packaging

   C. Hydraulic

      1) Relatively compact but requires return system and heating and circulation system for hydraulic fluid
      2) Requires monitoring of fluid content
      3) Fluid leak detection system required

2. Experience in Rocket Systems

   A. Pneumatic

      - Apollo, Delta

   B. Electrical

      - Transtage, Nerva, Apollo and OMS Actuator and He valves

   C. Hydraulic

      - Titan, Transtage, Delta

3. Contamination Sensitivity

   A. Pneumatic

      1) Seals susceptible to damage
      2) Small orifices can become clogged
      3) Dependent on cleanliness of service fluid
      4) Requires some electrical control

TABLE 3.4.3-2 (cont.)

Criterion

B. Electrical

1) Electrical section is sealed, problem free
2) Mechanical gearing could be jammed with contaminants
3) Contaminant caused short circuits

C. Hydraulic

1) Seals susceptible to damage
2) Small orifices can become clogged
3) Breakdown in fluid can lead to gums, varnishes that stop operation

4. Cycle Life

A. Pneumatic

1) Medium life due to dry system and lack of constant lubrication
2) Generally acceptable within OTV cycle life

B. Electrical

1) High life
2) Gear train wear and high speeds are limiting factors

C. Hydraulic

1) High life due to constant lubrication and warm operation
2) Contamination often limits life

5. Storage Life

A. Pneumatic

1) No particular problem
2) Limited by O-rings and seals
3) May require periodic stored gas servicing

B. Electrical

1) Longest storage life
2) Contact corrosion a concern with some designs

Criterion

C. Hydraulic

1) No particular problem
2) Limited by O-rings and seals

6. Response

A. Pneumatic

1) Compressible fluid may cause actuation lag
2) Erratic operation at low pressures

B. Electrical

1) Motor inertia must be overcome
2) Slower response at lower voltages
3) Response/weight tradeoff for most designs

C. Hydraulic

1) Incompressible fluid offers rapid response and best repeatability
2) Changes in pressure drop can change response time

7. Power Requirement

A. Pneumatic

1) Power required for pilot valves and servo valves only
2) Some heating power

B. Electrical

1) Power required for motors is generally not excessive
2) Wattage considered acceptable
3) Some heating power

C. Hydraulic

1) Power required for pilot valves, servo valves, heating system, and recirculating system. Constant current draw highest of the actuation methods

Table 3.4.3-2 (cont.)

Criterion

8. Adaptability

   A. Pneumatic

      1) Wattage requirements acceptable at voltage
      2) Regular servicing required

   B. Electrical

      1) Requires low voltage supply - generally available

   C. Hydraulic

      1) Requires hydraulic and low voltage supplies - hydraulic supply requires additional equipment
      2) Leaking fluid may pose a hazard

TABLE 3.4.3-3

VALVE ACTUATION METHODS CONCERNS

Concerns - Pneumatic

- Requires periodic servicing and gas pressure monitor system
- Poor lubricity
- Low bulk modulus, high compressibility (although relatively independent of temperature)
- Response time varies with number of cycles (gas source pressure decay)
- System is susceptible to leakage
- Cycle rate limited by supply volume
- Conclusion - conditionally acceptable for on-off type valves but not for modulating valves due to the continuous flow capacity needed for a dual-acting piston coupled with the compressibility of the gas - GH2 unacceptable due to safety concerns

Concerns - Hydraulic

- Need for a separate pressure source (i.e., pump)
- Thermal considerations regarding hydraulic fluid freezing in absence of heating system
- Propellant compatibility in case of leakage
- Valve response changes as a result of fluid temperature changes
  - Viscosity may change by a factor of 20,000 over temperature extremes of space
  - Range for organic fluids is -65°F to 900°F
  - Possible temperature buildup in a recirculating fluid
  - Bulk modulus varies greatly with temperature

Concerns - Electromechanical

- Larger, heavier valve than pneumatic or hydraulic
- Actuation time variation with temperature, voltage
- Electric power availability not defined for OTV
- Cycle life depends on geartrain configuration
- Conclusion - electromechanical actuation is acceptable

TABLE 3.4.3-4

VALVE ACTUATION TRADE STUDY SUMMARY

- Electrically actuated valves are acceptable for all engine requirements
- Valve development required for turbine bypass valve
  - No vendor catalog items meet requirements
  - Temperature and pressure range concern
  - Oxygen compatibility
- Good payoff for valve miniaturization, weight reduction
- Sensors should and can be designed into valves
- Some thermal conditioning will be required

TABLE 3.4.3-5

SHUT-OFF VALVES - TYPE SELECTION

• Angle Poppet	- Unacceptable	- Too heavy and bulky in sizes over 2 inches, high pressure drop
• Coaxial Poppet	- Unacceptable	- Internal actuation complicates design
• Sleeve	- Unacceptable	- Poor sealing properties, excessive leakage at sleeve seals
• Gate	- Unacceptable	- A lot of commercial experience, but not on rocket engines, too heavy, requires long stroke, high degree of design experience, low ΔP
• Ball	- Acceptable	- Design cycle life is limited due to seal rubbing
• Blade	- Acceptable	- Simple design, lightweight, low ΔP, design limited seal life due to seal rubbing is minimized due to low pressure application, little design experience
• Butterfly	- Unacceptable	- Simple design, high degree of design experience, low ΔP, may have poor low temp. sealing and cycle life concerns

TABLE 3.4.3-6

IGNITER AND TANK PRESSURIZATION VALVES - TYPE SELECTION

• Ball	- Unacceptable	- To heavy and bulky for small line sizes, slow response
• Butterfly	- Unacceptable	- Too heavy and bulky for small line sizes, slow response, questionable sealing
• Coaxial Poppet	- Unacceptable	- Internal actuation complicates design
• Gate	- Unacceptable	- Too heavy and bulky for small line sizes, slow response
• Sleeve	- Unacceptable	- Poor sealing properties
• Poppet	- Acceptable	- Tight shut-off, fast response, long seal life, simple design, design experience level is high

TABLE 3.4.3-7

MODULATING BYPASS VALVE SELECTION

- Fuel and Ox Turbine Bypass
- Heat Exchanger Bypass
- Fuel Regen Bypass

• Ball	- Unacceptable	- Poor throttleability due to seal damage with partially open high velocity flow
• Butterfly	- Unacceptable	- Difficult to throttle at high pressures, torque reversals during valve travel, high power requirements
• Gate	- Unaccedptable	- Poor throttleability rapid seat erosion in near closed position
• Pintle	- Acceptable	- Good throttling characteristics, precise control thru pintle shaping, low power due to balanced pintle, fast response due to short stroke, simple design concept

Main Problem Areas -

- Cryogenic effects on the Actuator
- Physically large size

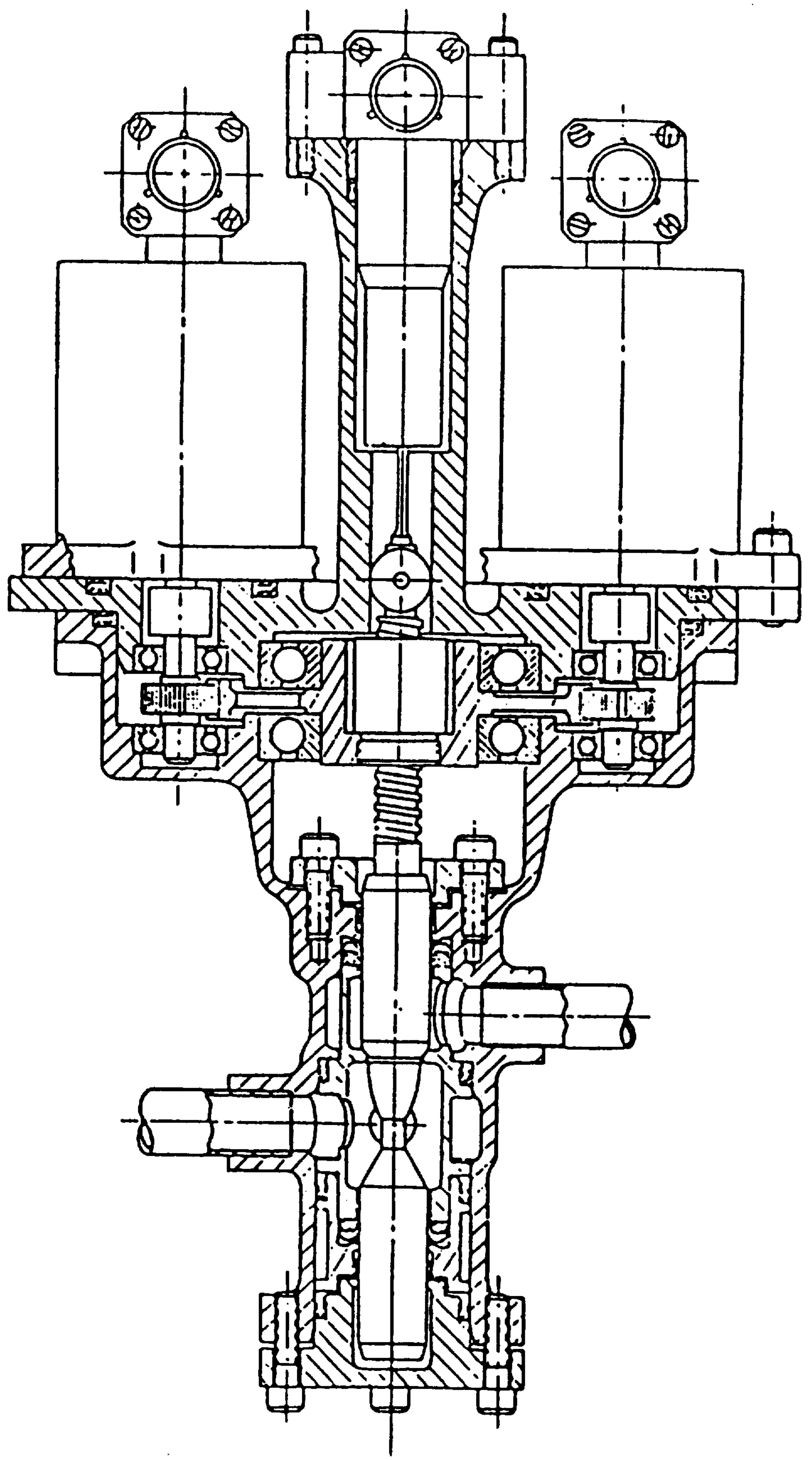


Figure 3.4.3-1 Turbine Bypass Valve Concept

TABLE 3.4.3-8

MODULATING HYDROGEN IDLE VALVE - TYPE SELECTION

• Butterfly	- Unacceptable	- To large and bulky for small line sizes, unacceptable throttling characteristics, lip seal wear may be minimized due to low velocity flow
• Gate	- Unacceptable	- Design experience level is low
• Poppet	- Acceptable	- Good throttling characteristics, precise control through poppet shaping, low power, fast response, easiest to design for fail-to-closed
• Ball	- Most Acceptable	- Has acceptable throttling and wear characteristics due to low velocity flow and non-continuous operation, lowest ΔP

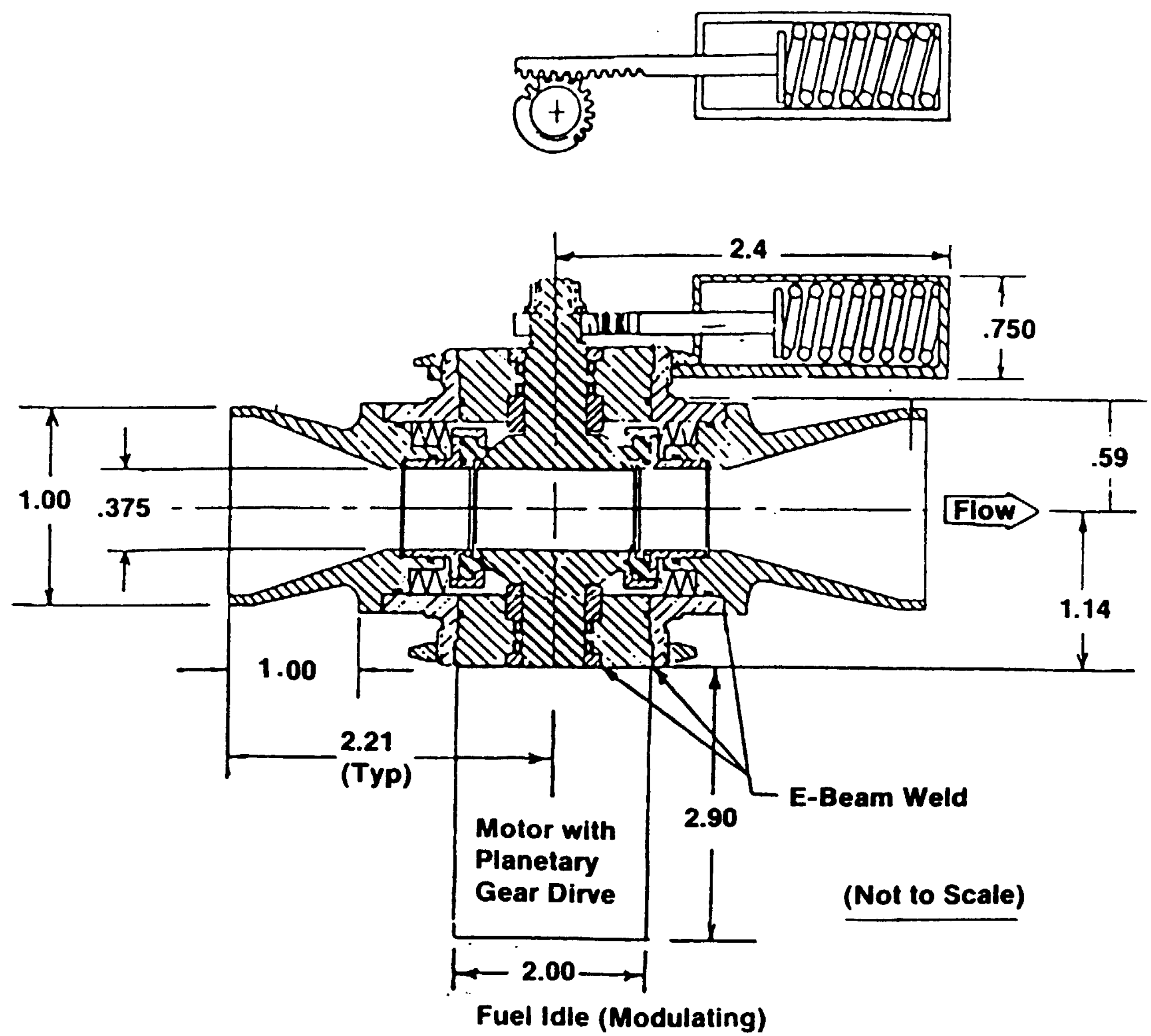


Figure 3.4.3-2 Hydrogen Idle Valve (Modulating)

increases above 500 psia. If a LOX/GH_2 heat exchanger (HEX) can accommodate the phase change without system fluctuations and film boiling inefficiencies the two valves can be removed. Such a design is possible, but until it is confirmed as troublefree the back pressure valves insure a smooth phase change. The trade study for these valves is given in Table 3.4.3-9. A poppet type valve with a spring preload to the high pressure drop position was selected. The back side of the valve is vented to a lower pressure portion of the system or, possibly, to vacuum. When the line pressure overcomes the spring pressure the valve moves to fully open. It is shown in Figure 3.4.3-3. At no time is the valve fully closed; there is always some flow thorugh the orificed area. It is also fully automatic with no electrical actuation required.

3.4.3.2 Valve Materials Selection

The criteria used for valve materials selection are:

- Non-reactive with oxygen (oxygen circuit)
- No hydrogen embrittlement (hydrogen circuit)
- Acceptable properties over the temperature range
- Ease of fabrication
- Mechanical properties
- Strength/weight ratio
- Dissimilar metals compatibility

The results of the materials selection are given in Table 3.4.3-10. As expected there is a heavy emphasis on materials compatible with the propellants. This list is subject to change as further work is done in the oxygen/materials compatibility area. An emphasis on weight reduction could also influence some of the selections.

3.4.4 Hydrogen Regenerator

A regenerator is a device that takes waste heat from an exiting stream in a system and transfers it to an entering stream where it can do useful work. Such devices usually add to the overall thermal efficiency of a system where heat is lost in an exiting stream. In a rocket engine where the production of high temperature, high pressure gases is the objective, the use of a regenerator is required only as a means of overcoming the bulk temperature rise of propellants at

TABLE 3.4.3-9

BACKPRESSURE VALVES, TYPE SELECTION

• Ball	- Unacceptable	- Non-adaptable to line pressure actuation
• Gate	- Unacceptable	- Non-adaptable to line pressure actuation
• Buttefly	- Unacceptable	- Susceptible to valve chatter resulting from dynamic flow oscillations
• Poppet	- Acceptable	- High level of design experience, minimal response time, simple concept, low ΔP, adaptable to pressure balance operation

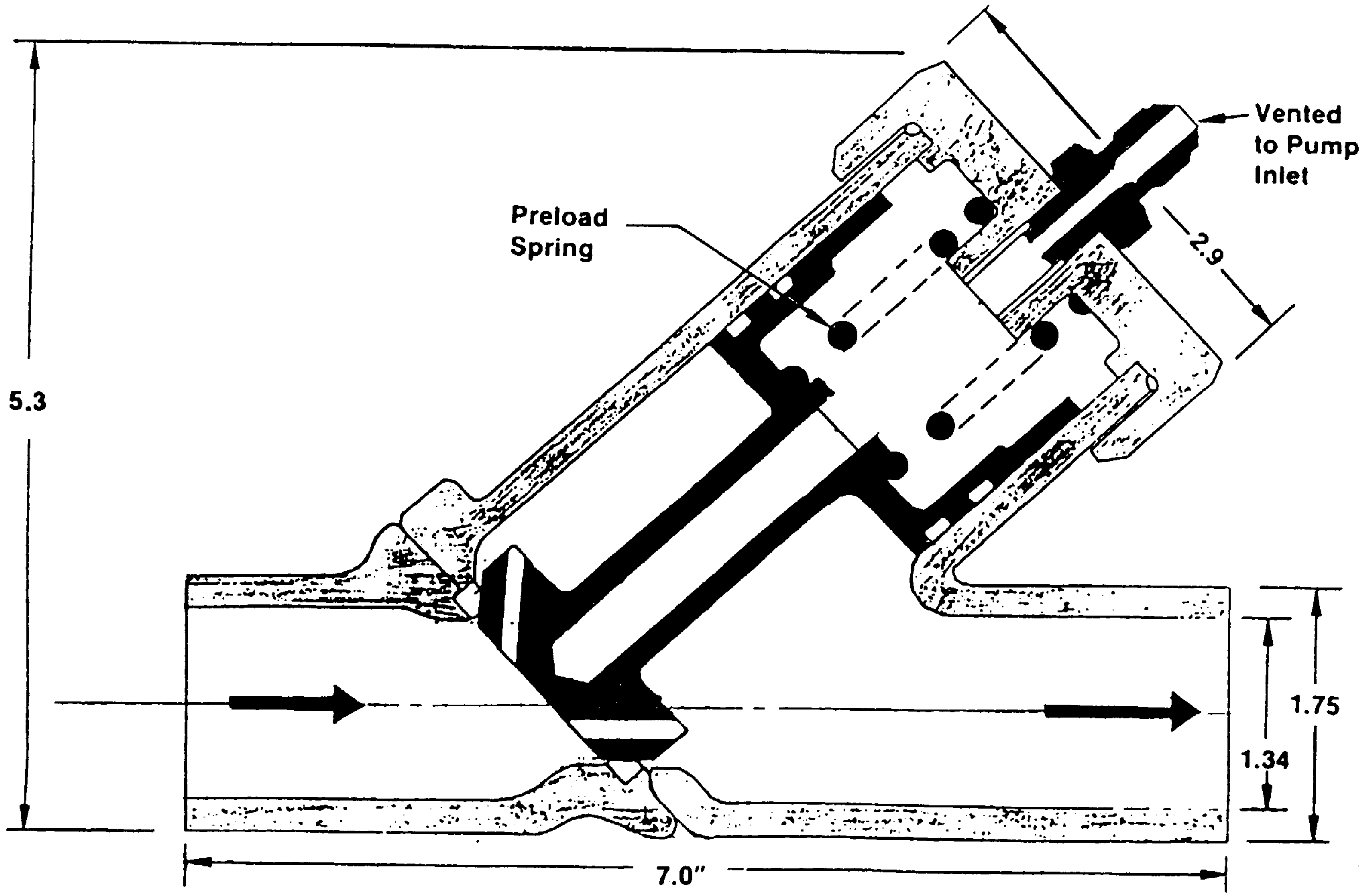


Figure 3.4.3-3 Line Pressure Actuated Back Pressure Valve Concept

Table 3.4.3-10 Preliminary Selection in Order of Preference for OTV Valve Components

Component	Material Candidates/ Max Use Temperature		Comments
Ox Turbine Bypass	1) Inconel 718	/1200°F	• Castable, but Needs Further Study
Temp (H/L) 400/400			
Press (H/L) 4937/330	2) Monel 400	/1000°F	• Good Impact Sensitivity
Hydrogen Turbine Bypass	1) Inconel 718	/1200°F	• Needs Further Study
Temp (H/L) 540/818	2) 347 CRES	/800°F	• Readily Fab.
Press (H/L) 4489/298	3) A286	/900°F	• Difficult to Weld
Heat Exchanger Bypass	1) 347 CRES	/800°F	• Greatest Fabricability
Temp (H/L) 440/753	2) Nitronic 50	/800°F	• Cannot be E.B Welded
Press (H/L) 2680/250	3) A-286	/900°F	• Difficult to Weld or Braze
	4) Inconel 718	/1200°F	• Needs Further Study
Regenerator Bypass	1) 6061-T6	/~200°F	• For Low Stress, Simple Config.
Temp (H/L) -340/-420	2) A356-T6	/~100°F	• For Low Stress, Complex Config.
Press (H/L) 5580/390	3) Ti-5AL-2.5n ELI	/~0°F	• Expansion Coefficient Similar to 440-C- S.S. (Common/Bearing Alloy)
Hydrogen Idle	1) 347CRES	/800°F	• Greatest Fabricability
Temp -360 to 540	2) Nitronic 50	/800°F	• Cannot be E.B. Welded
Press 0 to 200	3) A-286	/900°F	• Difficult to Weld and/or Braze
Hydrogen Main Shutoff	1) 6061-T6	/~200°F	• Could be Bar or Forging
Temp -422	2) A356-T6	/~100°F	
Press 25	3) Ti-5AL-2.5 Sn ELI	/~0°F	
Ox Main Shutoff	1) 6061-T6	/~200°F	• Could Be Bar or Forging
Temp -298	2) A356-T6	/~200°F	
Press 15	2) A356-T6	/~100°F	

Table 3.4.3-10 Preliminary Selection in Order of Preference for OTV Valve Components (Cont)

Component	Material Candidates/ Max Use Temperature	Comments
Hydrogen Backpressure Temp (H/L) 50 to 818 Press (H/L) 0 to 4889	1) Inconel 718 /1200°F 2) 347 CRES /800°F 3) A286 /900°F	
Ox Backpressure Temp -260 to 400 Press 0 to 4937	1) Inconel 718 /1200°F 2) Monel 400 /1000°F	• A Monel Would Simplify Line Welding
Hydrogen Tank Pressurization Temp 0 to 900 Press 0 to 3000	1) Inconel 718 /1200°F 2) 347 CRES /800°F 3) A286 /900°F	
Ox Tank Pressurization Temp 0 to 800 Press 0 to 3000	1) Inconel 718 /1200°F 2) Monel 400 /1000°F	
Fuel Igniter Temp -400 to 100 Press 0 to 6000	1) 347 CRES /800°F 2) Nitrogen 50 /800°F 3) A286 /900°F	• Easiest Fab
Ox Igniter Temp -400 to 100 Press 0 to 6000	1) Inconel 718 /1200°F 1) Inconel 718 /1200°F	• Good Impact Sensitivity • Good Impact Sensitivity

operating points below full thrust. With a regenerator the chamber can be physically smaller, thus reducing coolant hydrogen transit time and heat pickup during throttle down operation. Above 5600 lbf thrust the OTV engine needs more thermal energy than is available in thrust chamber and baffles. The regenerator bypass valve closes and hot turbine exit hydrogen is counterflowed with cold hydrogen from the pump. The thermal pickup is sufficient to raise the thrust to 7500 lbf before the hydrogen entering the regenerator is so hot that the chamber throat temperature approaches a structural limit.

3.4.4.1 Regenerator Design Concept

Aerojet has extended the platelet technology used for rocket engine injectors to applications where the small, intricate passageways possible with platelets are now used in a variety of heat exchanger applications. The result is a very compact heat exchanger of unusual shape and capable of very high heat transfer coefficients. An example of a recent design is given as Figure 3.4.4-1. This is a hydrogen regenerator used in the XLR-134 engine. The figure shows two flow circuits superimposed so that the channel interleaving and manifolds are evident. For hydrogen as a working fluid a zirconium copper or NASA-Z copper platelet structure is optimum. The platelets, which are thin plates with flow channels photoetched through the plate thickness, are assembled as a sandwich with thicker cover plates serving to closeout the top and bottom of the stack. The stack is held in place by guide pins until the assembly is diffusion bonded to form a completed unit. A leak check confirms that bonds were successful. In general, a bond joint should have the same strength as the parent metal.

3.4.4.2 Regenerator Design Discussion

Design objectives were to:

- Select a material that minimizes core weight and volume.
- Meet thermal performance criteria at the extreme engine operating points.
- Define an optimum channel geometry and length to meet the thermal performance criteria.
- Estimate weight and determine operating limits.

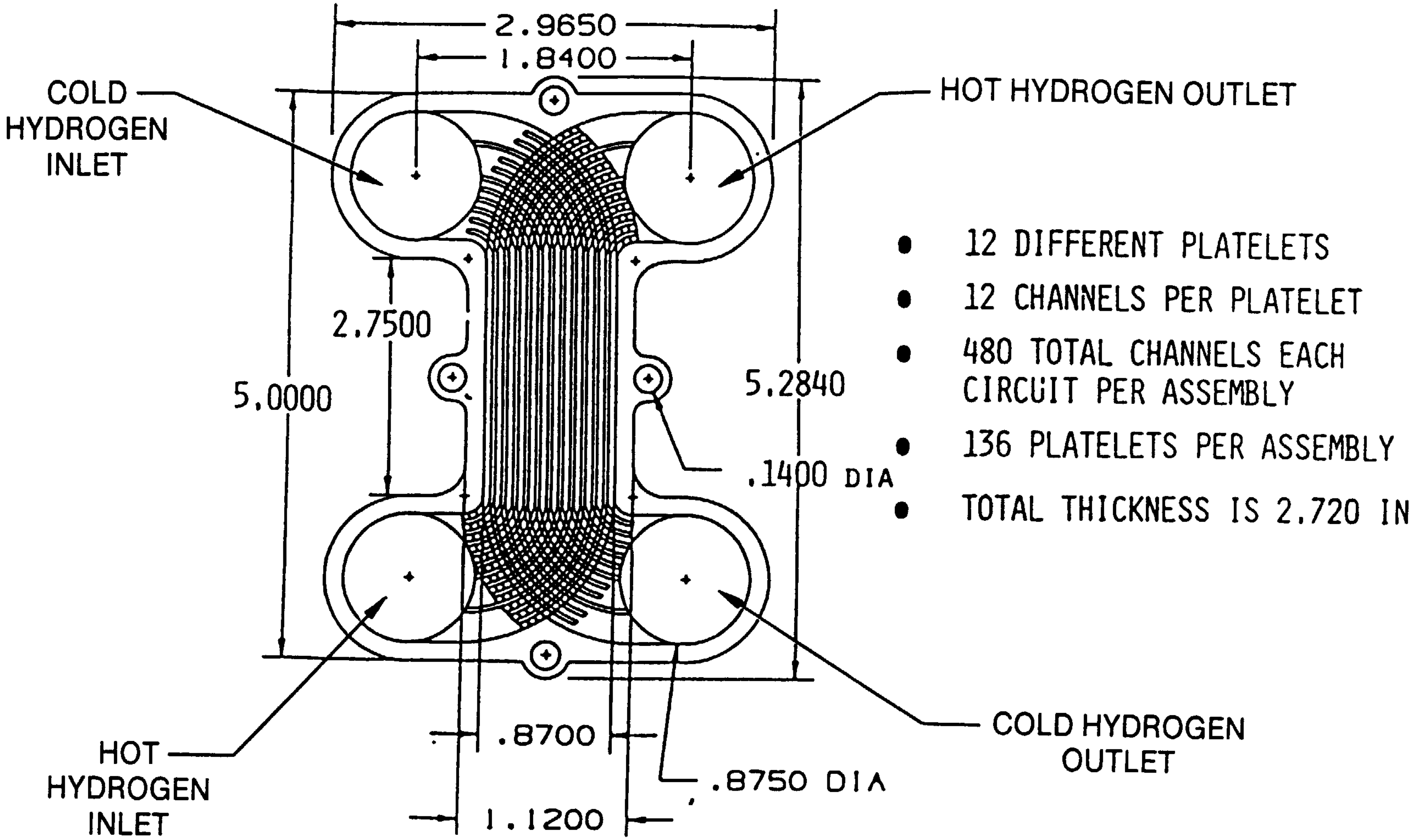


Figure 3.4.4-1 XLR-134 Regnerator is Typical of Platelet Heat Exchangers

Basic design criteria are listed in Table 3.4.4-1. Design conditions are given in Figure 3.4.4-2 along with a simplified circuit diagram. The wall thickness and land width calculations were based on preliminary stress calculations assuming material properties as follows:

ZrCu @ T = 800°F	Yield Strength = 8000 psi
Inconel 625 @ 800°F	Yield Strength = 42,000 psia

The regenerator weight and volume calculations are corrected by 15% to include mounting brackets, inlet-outlet weldable stubs and instrumentation. There is a design concept where the bypass valve would be a bolt on to the heat exchanger structure allowing a simple valve changeout without cutting any lines or opening a flange.

The regenerator design methodology is given in Figure 3.4.4-3. This methodology is used in the Aerojet computer code HEXSS.

Description of HEXSS

Both sizing and performance calculations for counterflow platelet heat exchangers are made with an Aerojet developed computer code. The code uses an iterative technique to solve the steady state energy and momentum equations for the cooled and heated fluid streams. Details of platelet geometry are accounted for in the evaluation of the overall conductance between the streams and in the determination of the core weight. Overall conductance is evaluated with knowledge of the fluid heat transfer film coefficients and a conductive fin/resistor network to model wall resistance.

In sizing calculations, parametric studies are conducted in which channel geometry, number of channels and channel length are systematically varied to establish a design. A design is established which meets all thermal/hydraulic requirements, while minimizing core weight and remaining within reasonably fabricable limits. The primary thermal/hydraulic requirements to be met are heat load, pressure drop and Mach number.

Once a design is established, off design performance of the heat exchanger is predicted by specifying inlet conditions while holding the geometry constant. An algorithm describing the sequence of operations followed in HEXSS when used in the performance prediction mode is as follows:

TABLE 3.4.4-1

REGENERATOR DESIGN CRITERIA

Operating Condition: Chamber Pressure = 2300 psia, Mixture Ratio = 7

Total Heat Transfer Rate	1108 Btu/sec
Hot Hydrogen Bypass	25%
Minimum Cold Hydrogen Exit Temperature	250°R
Maximum Cold Hydrogen Exit Temperature	<400°R
Maximum Cold Hydrogen Pressure Drop	50 psid
Maximum Channel Mach Number	0.3

For Operating Condition   Pc = 2300 psia   MR = 7

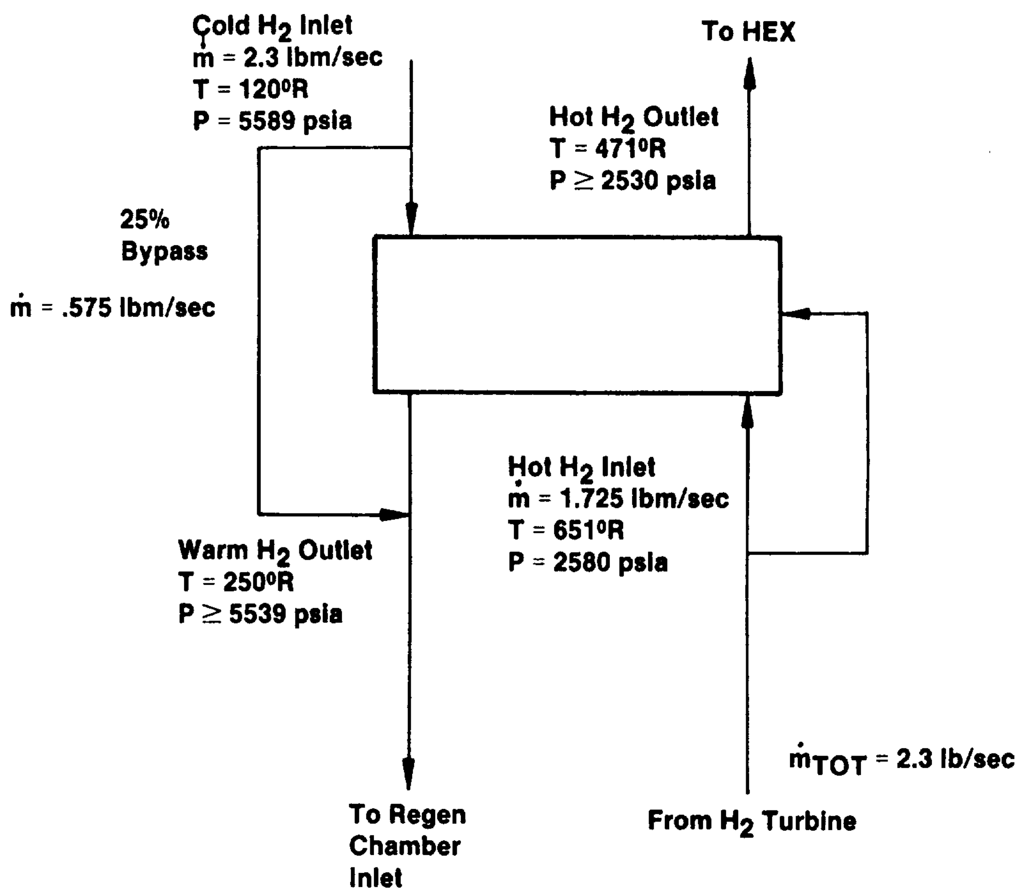


Figure 3.4.4-2 Regenerator Design Conditions

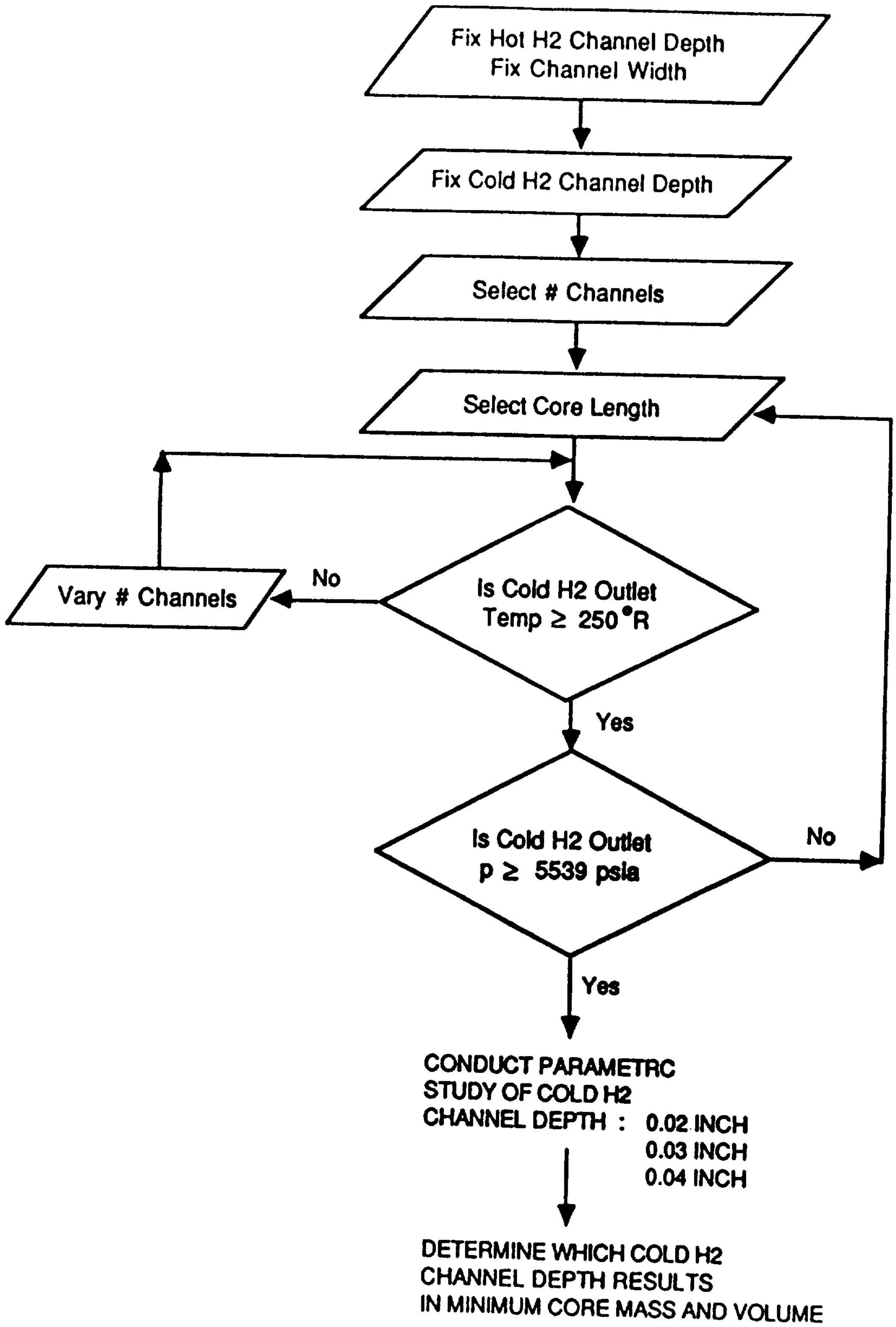


Figure 3.4.4-3 Regenerator Design Methodology

1. Input channel geometry and fluid inlet conditions.

2. Assume an initial distribution for hot gas/wall interface temperature from the inlet to exit.

3. Assume an initial distribution for hot gas bulk temperature and pressure from the inlet to exit.

4. Determine the axial distribution for the hot gas film coefficient using an appropriate correlation and the assumed bulk and hot gas/wall interface temperature distribution.

5. Begin calculations for cold gas side at the cold gas inlet.

6. Make an assumption for the temperature of the cold gas/wall interface temperature at the inlet.

7. Determine the cold gas film temperature at the current node based on bulk temperature and the assumed cold gas interface temperature.

8. Determine the cold gas film coefficient using an appropriate correlation and the current node film and bulk temperatures.

9. Determine the conductivity of the wall material at the average of the hot and cold gas wall/interface temperatures.

10. Calculate the overall heat transfer coefficient for the current node using the hot and cold gas film coefficients and evaluating the conductive resistance of the wall with a fin/resistor network.

11. Evaluate average heat flux for the node utilizing the overall heat transfer coefficient and the bulk temperature difference between the hot and cold gas.

12. Calculate hot and cold gas/wall interface temperature corresponding to overall heat flux, bulk temperature and film coefficient.

13. Check to see if calculated gas/wall interface temperatures agree with those assumed in determination of wall conductivity and cold gas film coefficient.

14. If interface temperatures are not in agreement update assumed wall temperature with a weighted average of calculated and assumed values; go to step 7.

15. Utilizing correct overall conductance for current node determine the amount of heat transferred to the cold stream as it flows through the node.

16. Evaluate the cold stream temperature in the node downstream from the current node knowing the increase in enthalpy corresponding to the heat transferred.

17. Evaluate the pressure drop through the current node and the pressure in the downstream node.

18. If the downstream node is at the cold gas exit then start recalculating hot gas temperature and pressure distribution, else downstream node becomes the current node and go to step 7.

19. Calculate hot gas film temperature at hot gas inlet, where hot gas temperature and pressure are known; previously calculated cold gas temperature, pressure and gas/wall interface temperatures are assumed correct.

20. Determine hot gas film coefficient for current node utilizing film and bulk temperature and an appropriate correlation.

21. Evaluate overall conductance and heat transferred from hot gas in current node using known bulk temperature difference, heat transfer coefficients and material conductivity.

22. Calculate hot gas bulk temperature in next node downstream corresponding to enthalpy drop through current node.

23. Determine pressure drop through current node and pressure in next node downstream.

24. If downstream node does not lie at hot gas exit then downstream node becomes the current node; go to step 20.

25. Check to see if previously assumed film and bulk temperature distributions for hot gas are in agreement with the newly calculated values.

26. If film and bulk temperature distributions have not converged then update hot gas bulk temperature distribution by forcing the heat added to the cold gas between a given node and the exit to be equal to the heat removed from the hot gas between the same axial locations. Go to step 4.

27. If bulk and film temperature distributions agree then calculate the total heat transferred to the cold stream and from the hot stream via the change in enthalpy between the inlet and exits.

28. If total heat transfer rates do not agree to within a specified amount then update hot gas bulk temperature as in step 26 and go to step 4.

29. Solution has been completed and fluid conditions throughout heat exchanger and other design information are written to output files.

Mechanical design consists of establishing the platelet patterns which, when stacked, will give the design configuration. An example is shown in Figure 3.4.4-4. Note that some platelets are through etched and some are only partially etched. This allows all channels in counterflowing streams to be separated by parent metal, not by a bonded surface. Leak paths will be to adjacent channels of the same stream, not to the counterflowing stream, if a diffusion bond is defective.

The HEXSS was run for both zirconium copper and Inconel 625. Results are given in Figures 3.4.4-5 and 3.4.4-6. The channel cross-section is shown in the upper left of the figures and the core length is available as curves with cold hydrogen outlet temperature and cold hydrogen outlet pressure as the ordinate. The core length of 3 inches in copper and 6.7 inches in Inconel is a reflection of the metal's thermal conductivities. This difference strongly favors a design in copper. The temperature approach (ΔT) for the counterflowing streams is given in Figure 3.4.4-7. The linear temperature plots show the uniform heat transfer properties of the platelet heat exchanger design.

With the zirconium copper selected for the regenerator, a further iteration was done to predict the bypass valve operating characteristic curve. This is presented in Figure 3.4.4-8. It shows the temperature to be very nearly a direct function of the amount bypassed which should simplify the valve design.

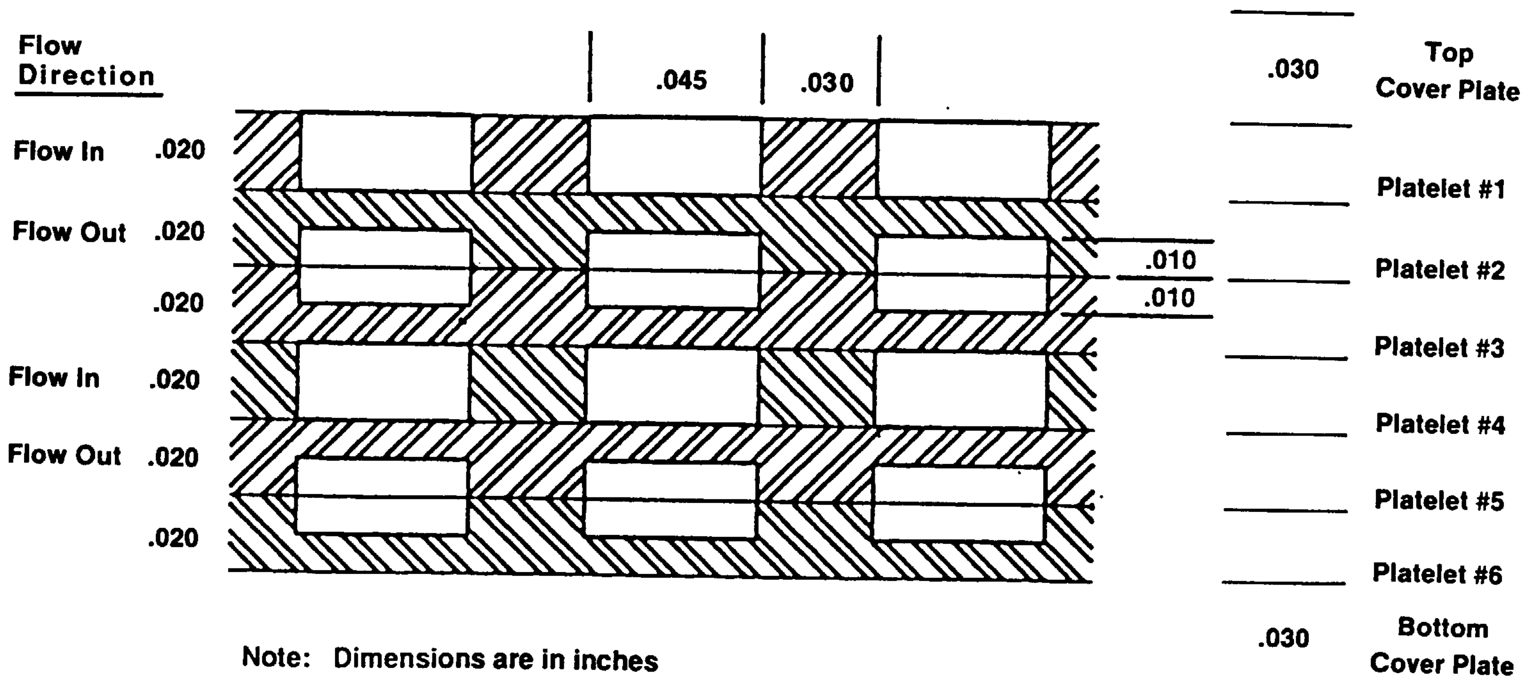
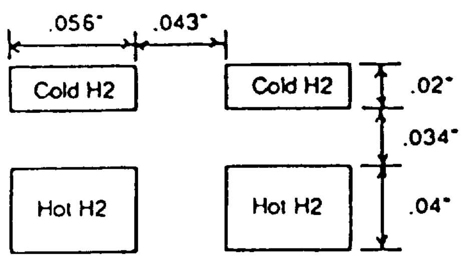


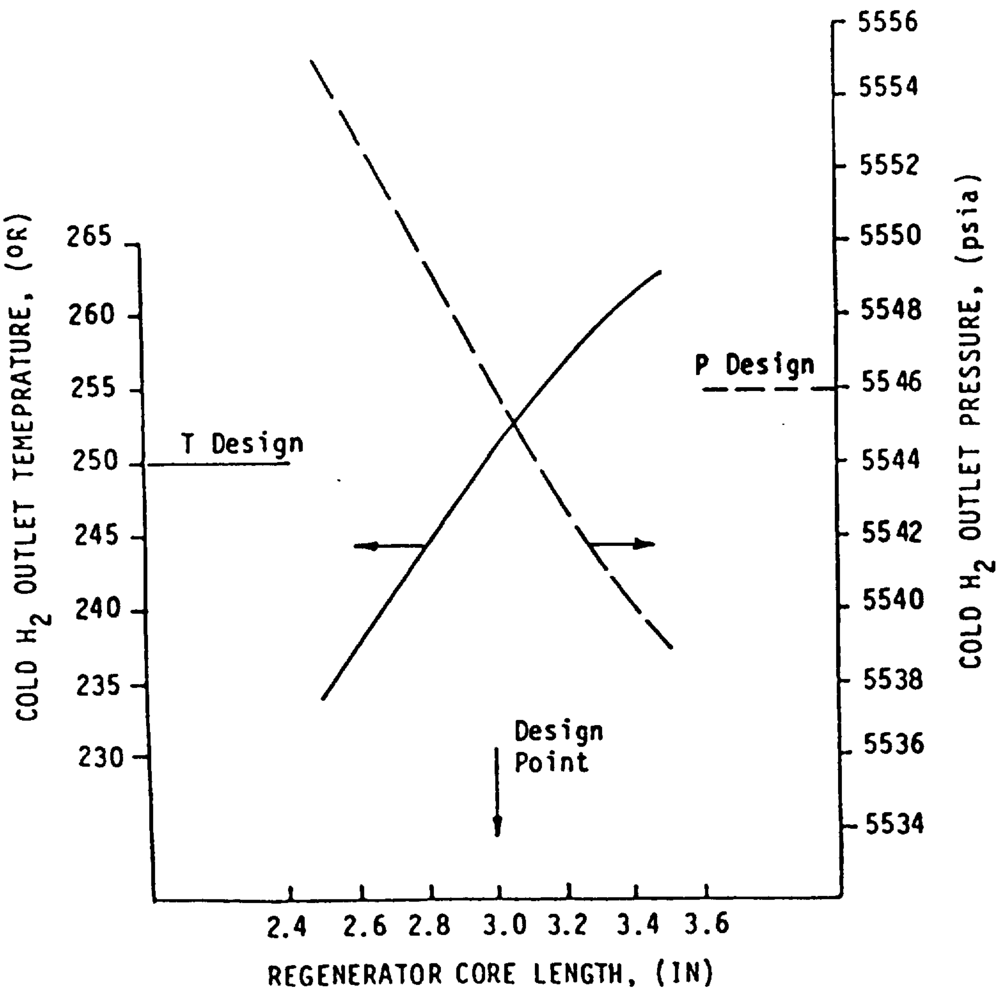
Figure 3.4.4-4 XLR-134 Regenerator Channel Details

ZrCu



	# Channels	$\dot{m}$ (lb/sec)	Tin (°R)	Tout (°R)	Pin (psia)	Pout (psia)
Cold H2	378	2.3	120	251	5589	5546
Hot H2	378	1.725	651	472	2580	2564

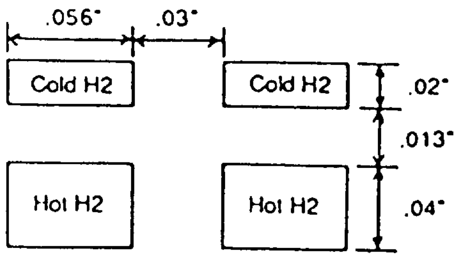
Core mass = 3.5 lb, Core volume = 14.7 in^3, Core length = 3.0 in



CONSTANT CHANNEL GEOMETRY AND TOTAL NUMBER OF CHANNELS
VARYING REGENERATOR CORE LENGTH

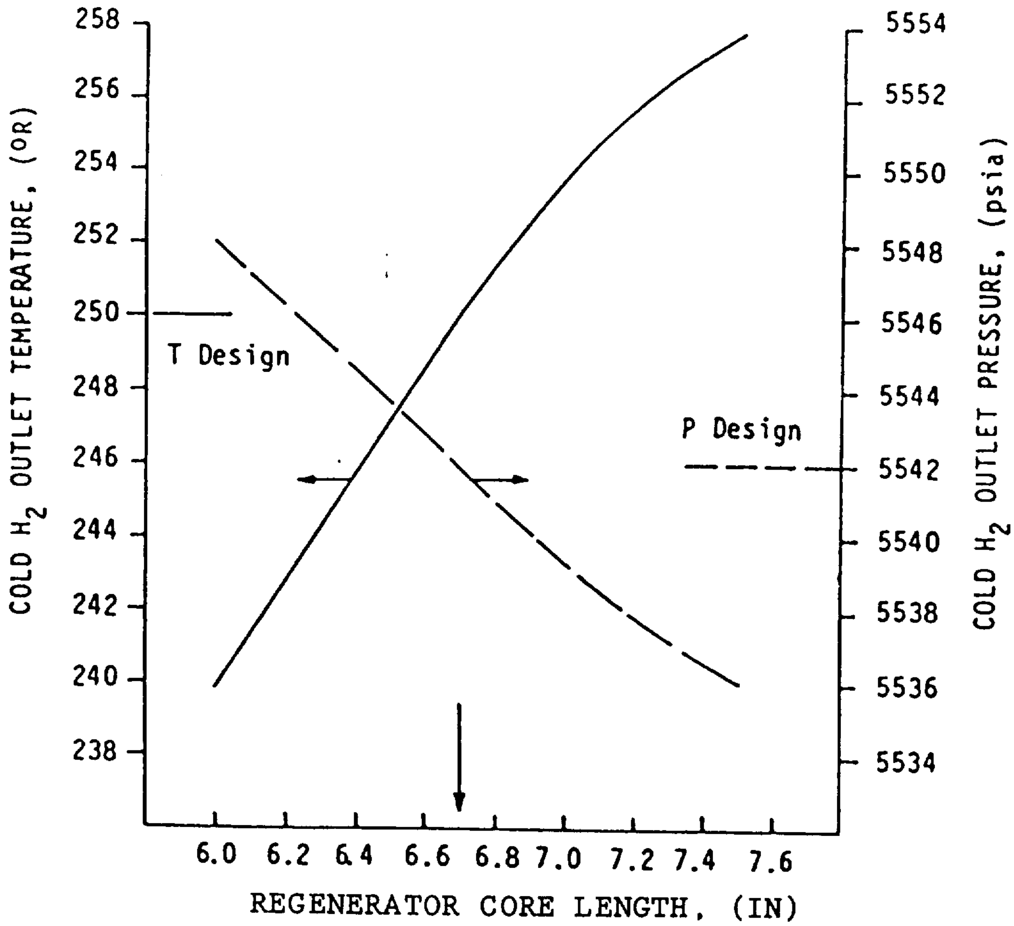
Figure 3.4.4-5 Regenerator Parametric Study Results

INCONEL 625



	# Channels	$\dot{m}$ (lb/sec)	Tin (°R)	Tout (°R)	Pin (psia)	Pout (psia)
Cold H2	540	2.3	120	250	5589	5542
Hot H2	540	1.725	651	472	2580	2562

Core mass = 4.6 lb, Core volume = 27.1 in^3, Core length = 6.7 in



CONSTANT CHANNEL GEOMETRY AND TOTAL NUMBER OF CHANNELS

VARYING REGENERATOR CORE LENGTH

* FOR PRELIMINARY REFERENCE ONLY - HEAT EXCHANGER MODEL MODIFICATION IN PROGRESS TO ACCOMODATE INCONEL 625

Figure 3.4.4-6 Regenerator Preliminary Parametric Results

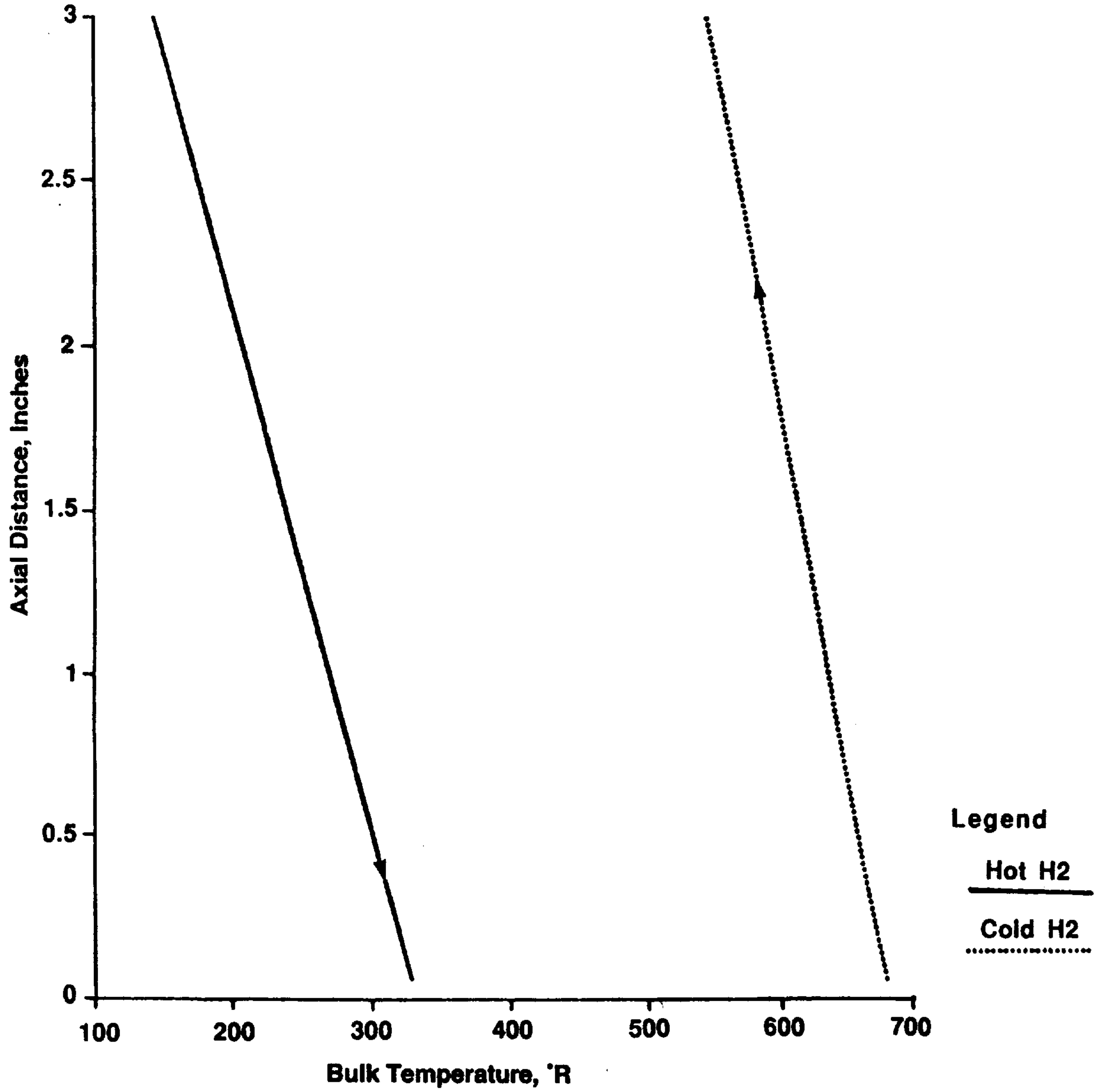
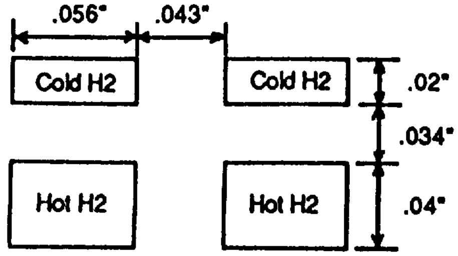


Figure 3.4.4-7 OTV Regenerator Fluid Temperatures Counter Flow Configuration

ZrCu   Pc = 2300 psia,   MR = 7



	# Channels	$\dot{m}_{TOT}$ (lb/sec)	Tin (°R)	Pin (psia)
Cold H2	378	2.3	120	5589
Hot H2	378	2.3	651	2580

Core length = 3.0 in

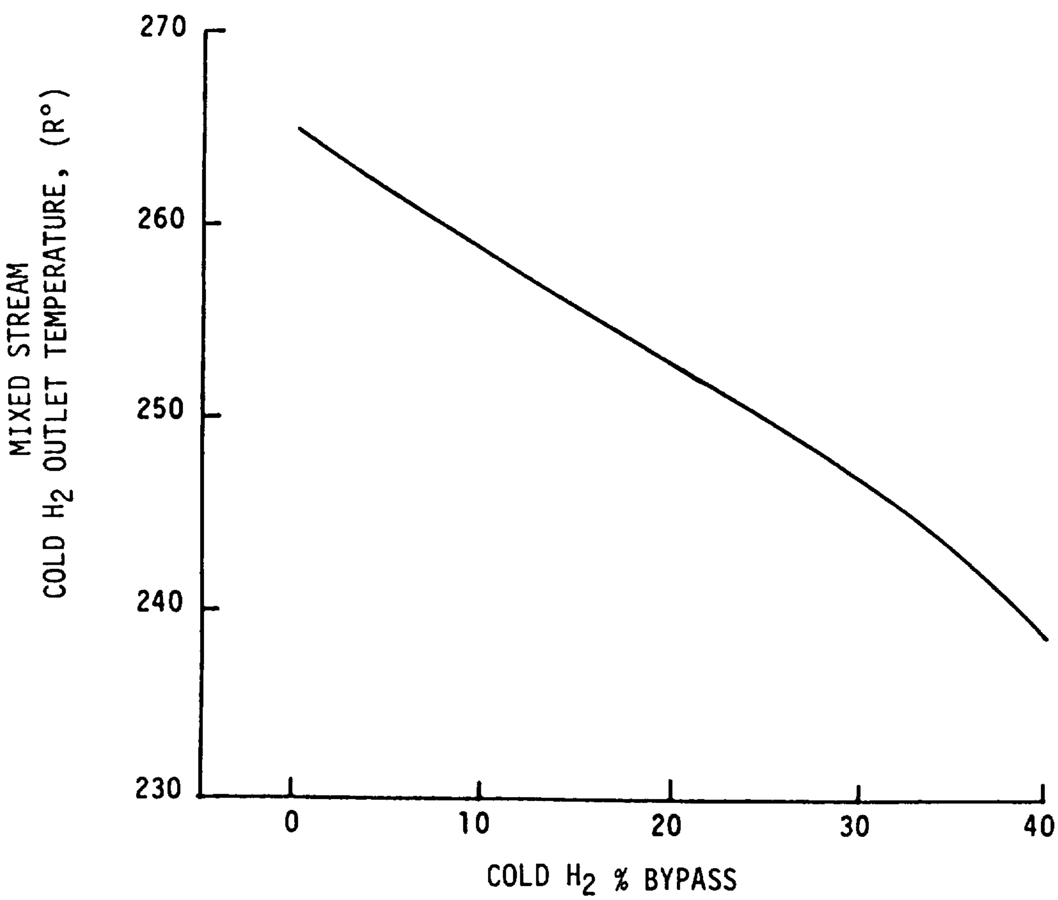


Figure 3.4.4-8 Regenerator Bypass Valve Operation

3.4.4.3 Design Summary

The parameters for the regenerator design are given in Figure 3.4.4-9. Results are summarized in Table 3.4.4- 2. Note that the core mass is only 3.5 lbm with a completed design weight of 4.1 lbm. This is a very effective component considering that it allows a thrust chamber with a thermal design established for 4600 lbf of thrust to operate at 7500 lbf of thrust using hydrogen turbine waste heat.

3.4.5 Liquid Oxygen/Gaseous Hydrogen Heat Exchanger Design

The LO_2/GH_2 heat exchanger (HEX) is a component necessary for the dual expander cycle. Warm (400°F) oxygen is needed for the turbine of the oxygen turbopump. Heating is done in two stages: 1) in the HEX where approximately two thirds of the enthalpy change occurs, and 2) in the oxygen cooled nozzle where the balance of the heating occurs. The basic temperature control is done by the HEX bypass valve operating in a closed loop using the temperature at the oxygen turbine inlet for a reference. The flow path for the regenerator and HEX circuits is shown in Figure 3.4.5-1. The hot hydrogen from the TPA turbine outlet flows in series from the HEX through the regenerator and then to the injector. A subsidiary benefit of this design is that the hydrogen entering the injector is relatively cool and dense allowing for a more compact injector design.

3.4.5.1 HEX Design Discussion

The objectives for the HEX design were to:

- Determine channel geometry
- Determine HEX weight and dimensions
- Determine a design solution for the oxygen phase change

Two cases were considered for separate design consideration: the supercritical oxygen property range, and the two phase oxygen property range. The supercritical case was readily solved. The two phase case was not solvable within the hour allocation for the task. This was not an unexpected result as the NASA LeRC had funded two HEX development programs for the RL-10 engine, and the first was unsuccessful. The second used an unusual heat exchanger flow circuit with a very small ΔT in the region where the phase change takes place. This controlled the

TOTAL H2/H2 REGENERATOR LENGTH	3.0 in
NUMBER OF HOT HYDROGEN CHANNELS	378
NUMBER OF COLD HYDROGEN CHANNELS	378
CHANNEL WIDTH	0.056 in
DISTANCE BETWEEN LIKE CHANNELS (LANDS)	0.043 in
WALL THICKNESS BETWEEN UNLIKE CHANNELS	0.034 in
COLD HYDROGEN CHANNEL DEPTH	0.02 in
HOT HYDROGEN CHANNEL DEPTH	0.04 in
TOTAL REGENERATOR CORE VOLUME	14.7 in^3
TOTAL REGENERATOR CORE WEIGHT	3.51 lb
MATERIAL	ZrCu

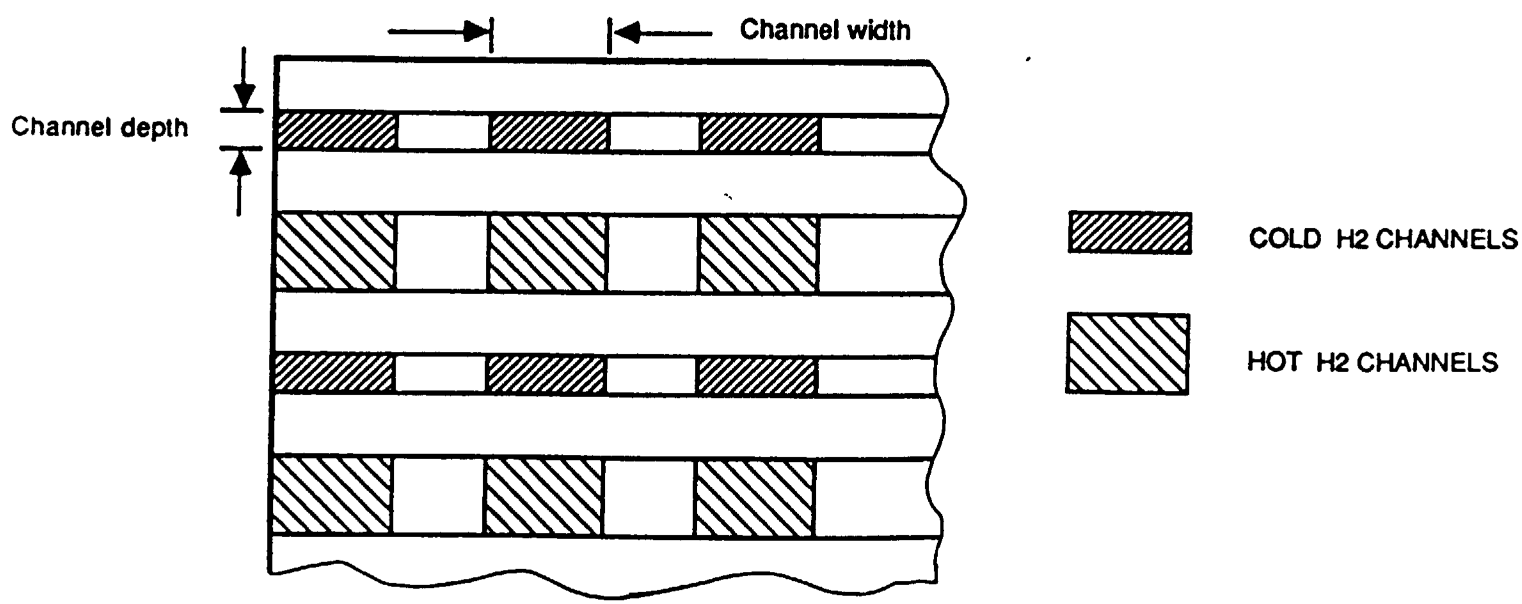
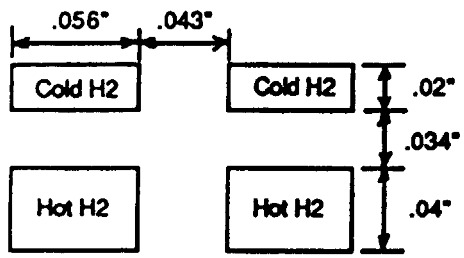


Figure 3.4.4-9 Regenerator Design Parameters

Table 3.4.4-2 Summary of Results for Regenerator



ZrCu

	# Channels	Pc = 2300 Tin (°R)	Pc = 2300 Pin (psia)	Pc = 2075 Tin (°R)	Pc = 2075 Pin (psia)	Pc = 200 Tin (°R)	Pc = 200 Pin (psia)
Cold H2	378	120	5589	120	4898	40	398
Hot H2	378	651	2580	644	2325	1035	225

Core mass = 3.5 lb, Core volume = 14.7 in^3, Core length = 3.0 in

	Pc (psia)	MR	Q TOTAL REQ'D (Btu/sec)	$\dot{W}$ COLD H2 (lb/sec)	ΔT COLD H2 (°R)	ΔP COLD H2 (psid)	$\dot{W}_{TOT}$ HOT H2 (lb/sec)	BYPASS HOT H2 (%)	ΔT HOT H2 (°R)	ΔP HOT H2 (psid)	ΔT HOT H2 MIXTURE (°R)
DESIGN POINT	2300	7	1108	2.3	131	43	2.3	25	179	18	136
OFF-DESIGN	2075	7	1001	2.6	105	55	2.6	48	214	10	114
	200	5	186	0.3	175	3	0.3	71	659	1	194

* ALL RESULTS BASED ON POWER BALANCE RECEIVED OCTOBER 1987

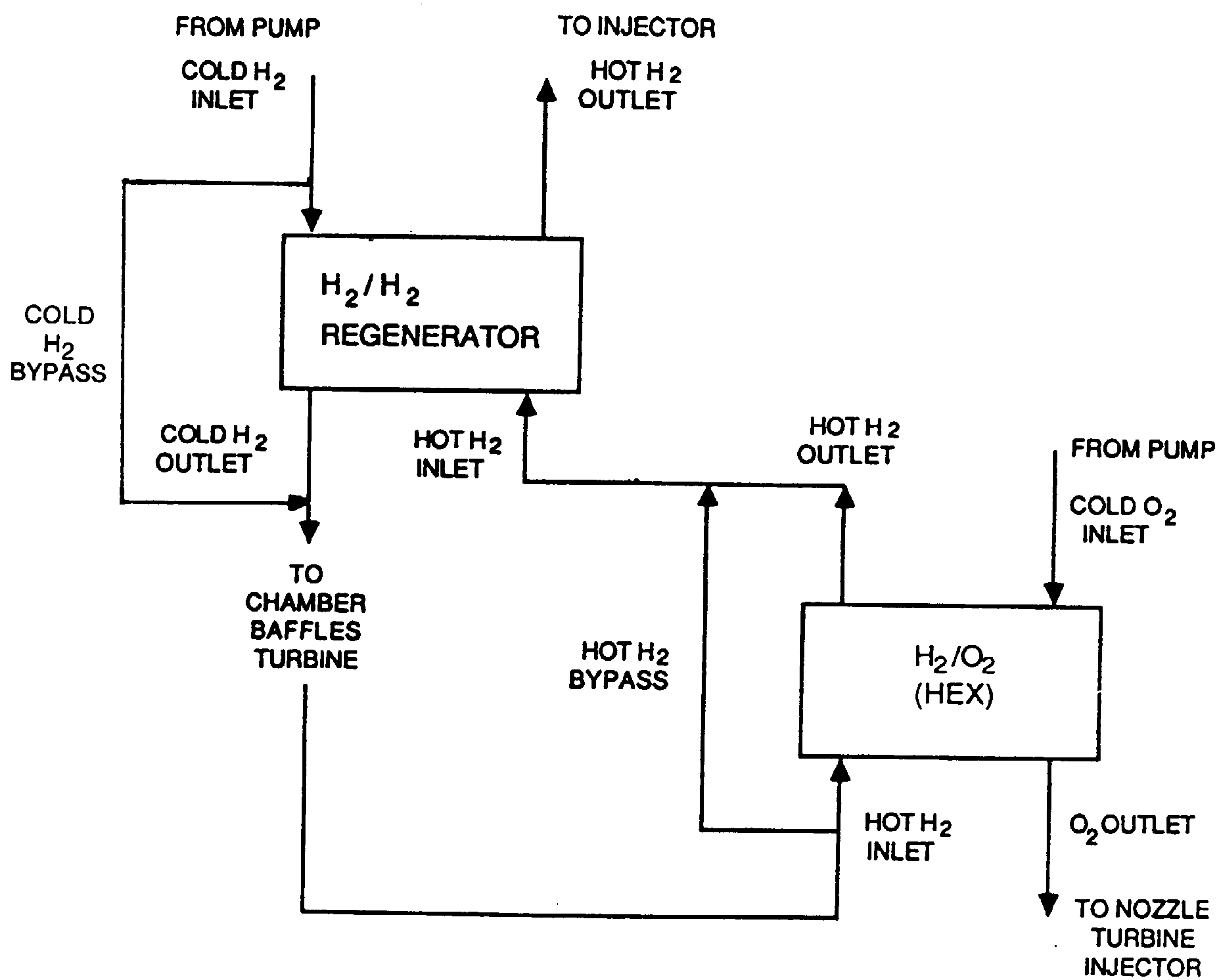


Figure 3.4.5-1 H_2/H_2 Regenerator - H_2/O_2 HEX

propensity of oxygen to enter a film boiling regime at high heat fluxes. It did add to the heat transfer area and increased the weight of the design. The Aerojet intent was to keep to minimum size and weight in the design. The options that were explored were:

- Decrease the heat flux by using a low conductivity material for the heat exchanger.

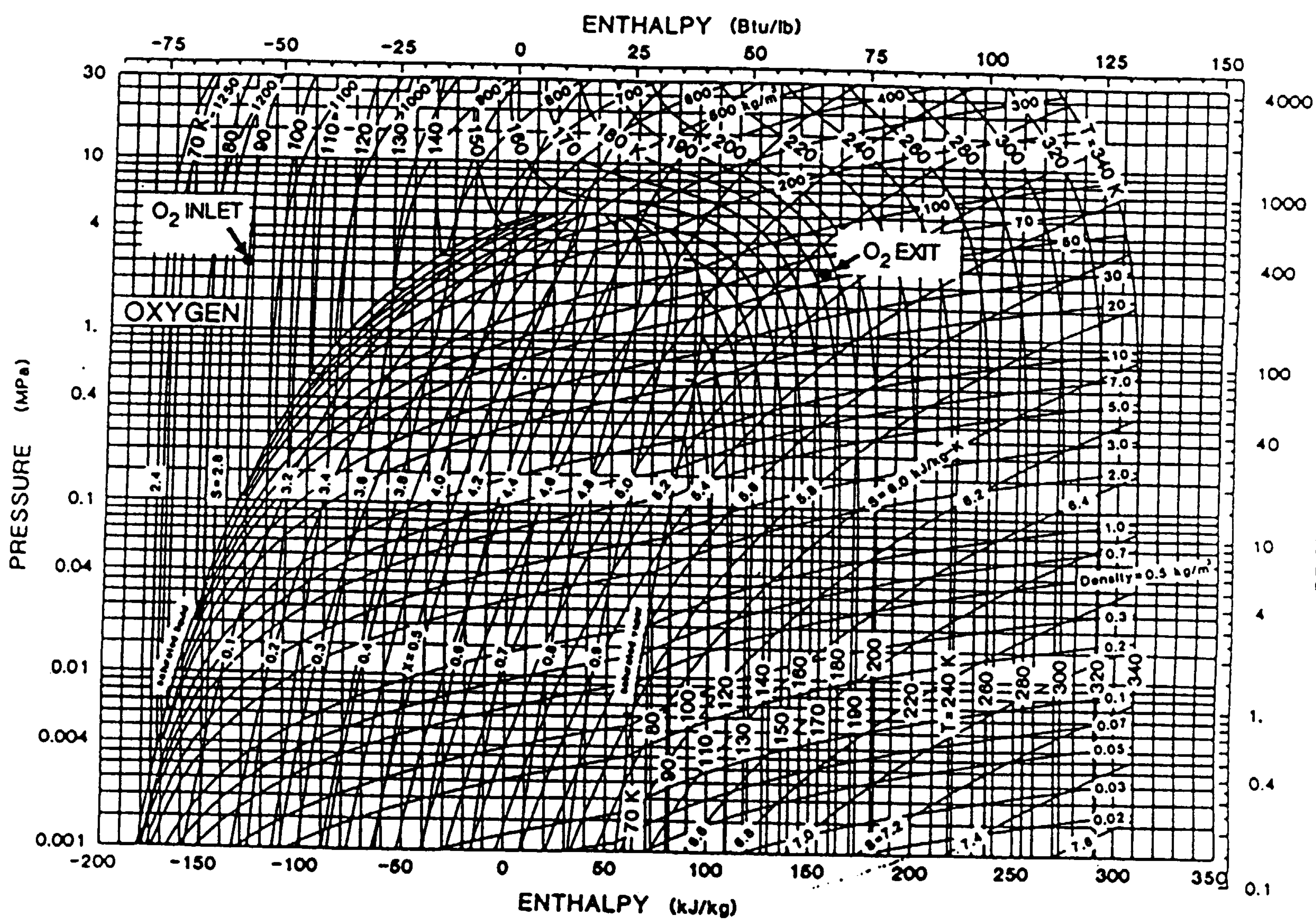
- Accept film boiling but negate its effects by using an induced turbulence mixing technique.

The monel metals were considered for HEX material but did not solve the film boiling condition and the HEXSS computer program could not run with the great variation in oxygen thermodynamic properties in the two phase region. The oxygen inlet condition at a chamber pressure of 200 psia is shown on Figure 3.4.5-2 along with the desired oxygen exit condition from the HEX. The oxygen enthalpy change through the system occurs as it traverses the two phase "dome" region. Properties in this region are highly variable and may be impossible to model as a liquid to gas phase change is a chaotic process at non-equilibrium conditions. A less than ideal solution is to raise the system pressure above the critical point. This is proposed as the function of the back pressure valve used in the oxygen circuit. A similar valve in the hydrogen circuit is used to balance the pressure between the two circuits; there is no corresponding problem in the hydrogen phase change. A more elegant solution would be to design the HEX so that it handles the phase change.

Figure 3.4.5-3 illustrates the film boiling condition as a problem for the design code. Homogeneous flow is easily modeled, but in film boiling the heat transfer surfaces "see" only a vapor film with the saturated liquid confined to a flow path in the center of the channel. At high fluid velocities the liquid can pass right through the heat exchanger without picking up the necessary heat of vaporization. Effectively, film boiling lowers the heat transfer coefficient. A potential design solution for a platelet heat exchanger is shown in Figure 3.4.5-4. The oxygen channel geometry is modified to force turbulent mixing of the fluid. This could break up the entrained saturated liquid slugs and improve the heat transfer. Another channel configuration would incorporate in-channel fins of an aerodynamic shape that would generate tip vortexes. The resulting vortexes could be very effective turbulent mixers. This concept would be explored in any detail design phase for a HEX.

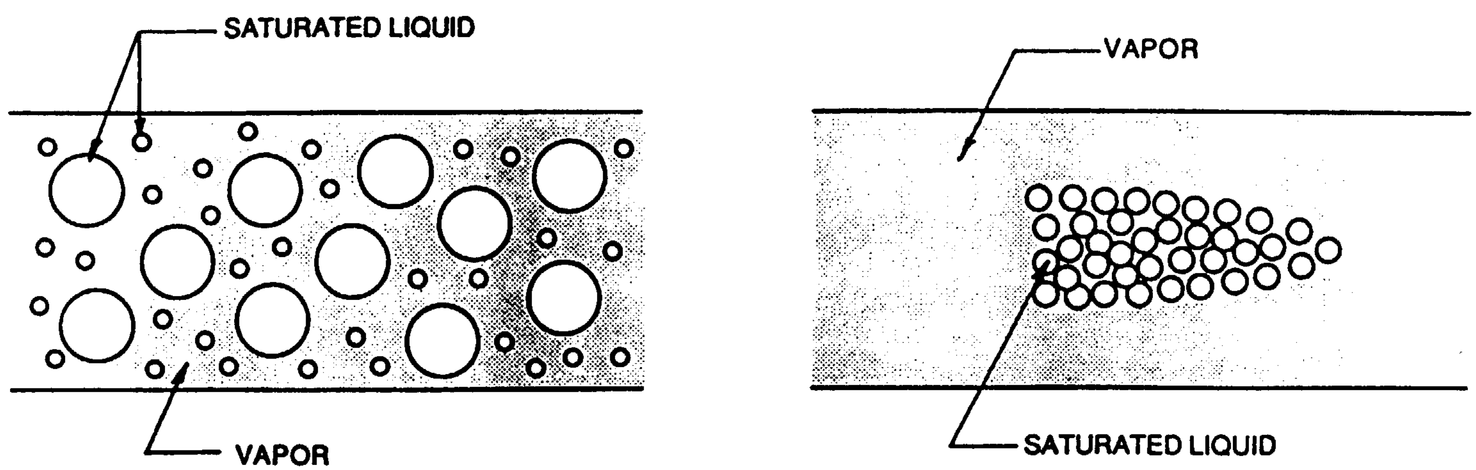
The design methodology for the HEX is given in Figure 3.4.5- 5. The HEXSS code was adapted to run this program logic. The materials properties input was based on

OXYGEN PHASE CHANGE Pc = 200 psia MR = 5



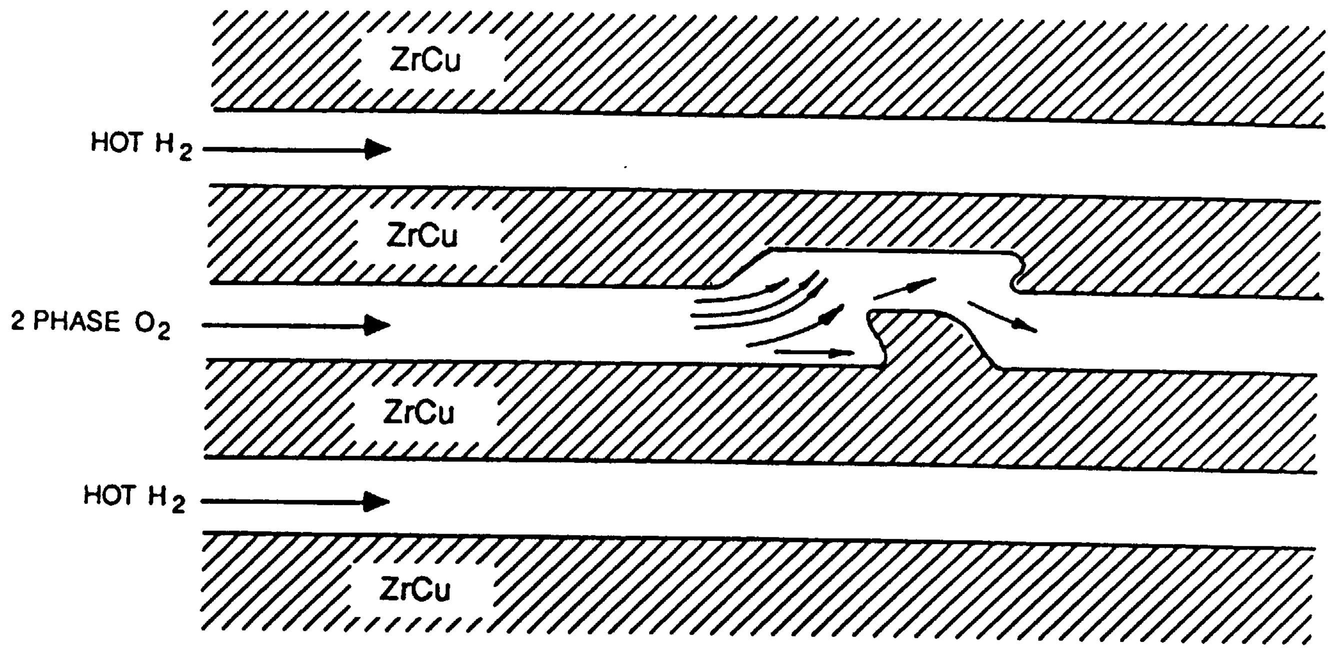
PROBLEM : 1. HEAT EXCHANGER MODEL INTERPOLATES PROPERTIES ACROSS PHASE CHANGE
2. HEAT EXCHANGER MODEL HAS NOT BEEN SUFFICIENTLY CHECKED FOR TWO PHASE FLOW

Figure 3.4.5-2 Two - Phase O_2 Case



PROBLEM : HEAT EXCHANGER MODEL CONSIDERS ONLY HOMOGENEOUS FLOW.

Figure 3.4.5-3 Film Boiling Potential



DESIGN OBSTRUCTIONS IN O2 FLOW PATH TO MAINTAIN A HOMOGENEOUS MIXTURE AND ELIMINATE FILM BOILING POTENTIAL

Figure 3.4.5-4 Design Approach for Two - Phase O_2 Case

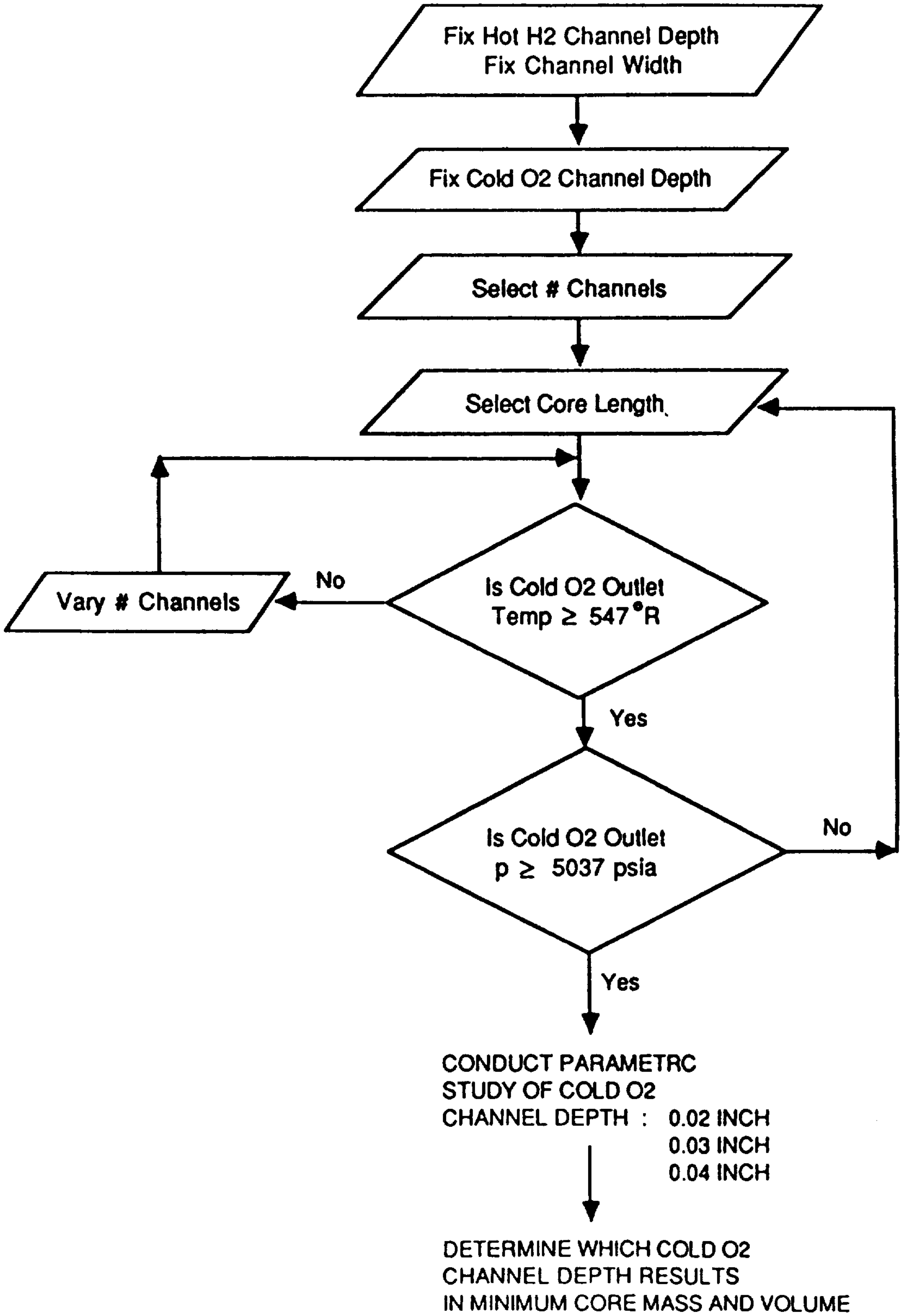


Figure 3.4.5-5 Supercritical O_2 Case HEX Design Methodology

zirconium copper at 900°F as worst case. Design conditions are presented in Figure 3.4.5-6. This is a worst case condition based on engine overthrust operation to 2300 psia at MR = 7. The O_2 outlet temperature was based on a predicted enthalpy change in the oxygen cooled nozzle. With that heat addition the oxygen would enter the TPA turbine at 400°F.

The supercritical case parametric study yielded results summarized on Figure 3.4.5-7. Note that channel geometry is given in the upper left of the figure. The HEX design point yielded a core length of 11.6 inches. This reflects the poor heat transfer characteristics of oxygen compared to hydrogen and also the greater temperature change as compared to the regenerator.

The two phase design conditions are summarized in Figure 3.4.5-8. A confirmation that the supercritical design would also meet the two phase design conditions was beyond the scope of this task. It was evident that the HEXSS code would require substantial modification to handle this condition.

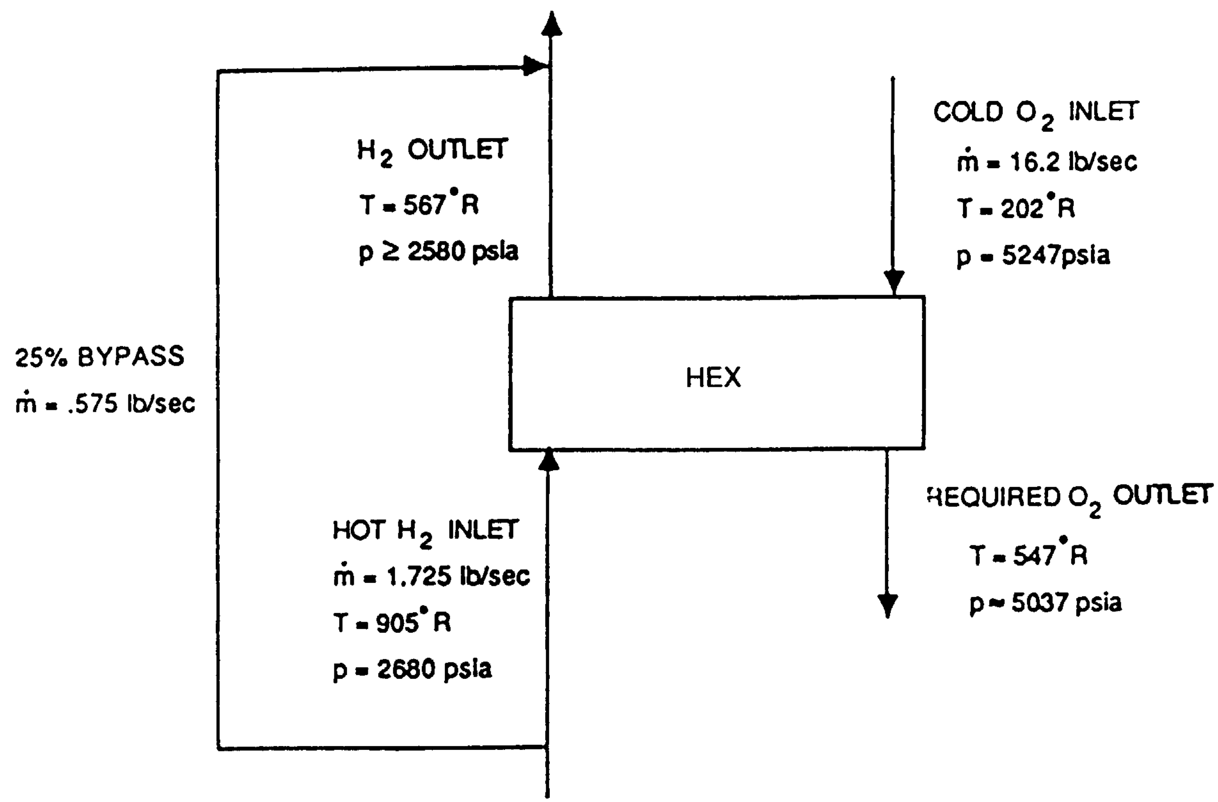
3.4.5.3 Design Results

The supercritical case results are presented in Figure 3.4.5-9. Note that core mass is 16.6 lbm with a finished weight estimate of 19.1 lbm. The construction material is zirconium copper. The assumption is that this supercritical design can handle the two phase flow condition with some additional work in devising a high mixing rate channel configuration. Physically the HEX is still a small component and was readily packaged in the component layout work.

3.4.6 Oxygen Cooled Nozzle Design

The section of nozzle from the oxygen inlet manifold at area ratio 35 to the envelope limit of 60 inches must be regeneratively cooled. The cooling requirement is also an opportunity to add some heat to the oxygen and reduce the size of the LO_2/GH_2 heat exchanger (HEX). Initial calculations show that one third of the necessary enthalpy change for the oxygen stream can be acquired in cooling the nozzle. Also, the heat transfer rates are well within the cooling capability of oxygen. In addition there may actually be some weight savings as the oxygen operates at a lower pressure than would a hydrogen coolant and the structure can be designed with thinner walls.

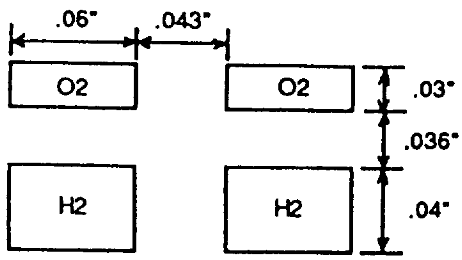
FOR SUPERCRITICAL CASE Pc = 2300 psia MR = 7



DESIGN CRITERIA : O_2 OUTLET TEMPERATURE 547°R
MINIMIZE PRESSURE DROP
MAXIMUM CHANNEL MACH NUMBER 0.3

Figure 3.4.5-6 LO_2/GH_2 HEX Design Conditions

ZrCu Pc = 2300 psia, MR = 7



	# Channels	$\dot{m}$ (lb/sec)	Tin (°R)	Tout (°R)	Pin (psia)	Pout (psia)
Cold O2	414	16.2	202	547	5247	5052
Hot H2	414	1.725	905	568	2680	2610

Core mass = 16.6 lb, Core volume = 71.5 in^3, Core length = 11.6 in

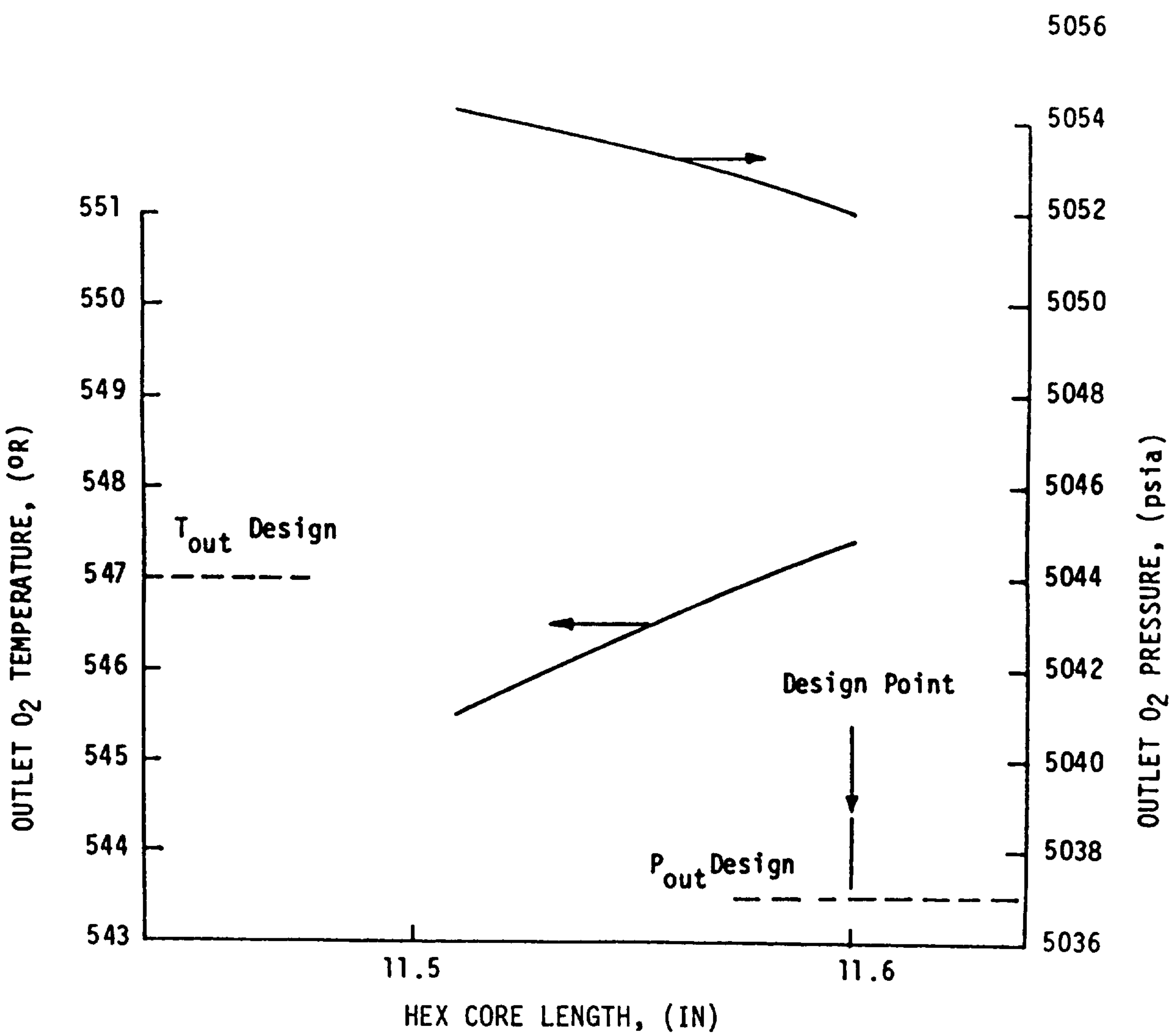
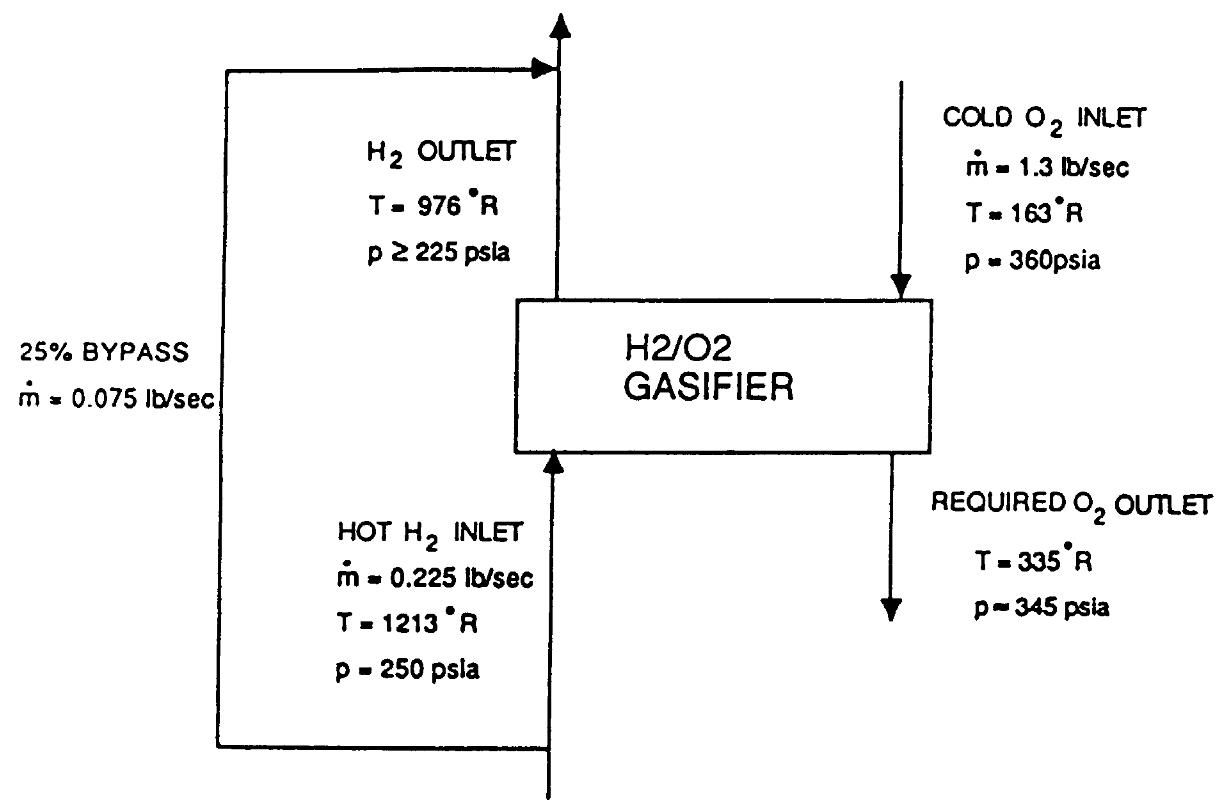


Figure 3.4.5-7 HEX Parametric Study

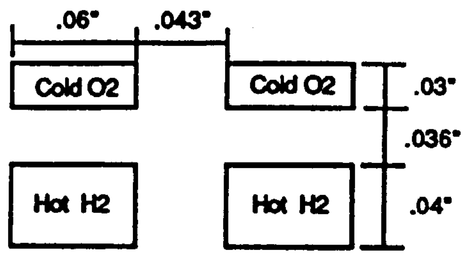
FOR TWO-PHASE O2 CASE   Pc = 200 psia   MR = 5



DESIGN CRITERIA: O_2 OUTLET TEMPERATURE 335°R
MINIMIZE PRESSURE DROP
FILM BOILING IS UNACCEPTABLE

Figure 3.4.5-8 LO_2/GH_2 HEX Design Conditions

Pc = 2300 psia MR =7



	# Channels	Tin (°R)	Pin (psia)
Cold O2	414	202	5247
Hot H2	414	905	2680

Core mass = 16.6 lb, Core volume = 71.5 in^3, Core length = 11.6 in

MATERIAL	Q TOTAL REQ'D (Btu/sec)	w COLD O2 (lb/sec)	ΔT O2 (°R)	ΔP O2 (psid)	MAX O2 MACH #	WTOT HOT H2 (lb/sec)	BYPASS HOT H2 (%)	ΔT HOT H2 (°R)	ΔP HOT H2 (psid)	ΔT HOT H2 MIXTURE (°R)	MAX H2 MACH #
ZrCu	2059	16.2	345	195	0.11	2.3	25	337	70	253	0.081

Figure 3.4.5-9 Summary of Results for HEX Supercritical O_2 Case

3.4.6.1 Cooled Nozzle Design Concepts

The nozzle envelope dimensions are given in Figure 3.4.6-1. The oxygen inlet in all concepts is at an area ratio of 35. This inlet manifold is assumed to be integral with or welded to the hydrogen inlet manifold. No resources were available for an in-depth design of the manifolds. The hydrogen enters at a nominal area ratio of 28. Since the two streams must be physically separate the manifold design has to assure good heat transfer at the manifold but safe separation of the two streams. For the oxygen flow beyond the manifold several options are available. All had to meet the design requirements given in Table 3.4.6-1 for both flow circuit design and material properties.

Candidate materials for the nozzle were evaluated based on oxygen compatibility, low density (low weight), and high thermal conductivity. Because of the poor coolant properties of oxygen, a material which conducts heat readily is desirable to enhance the cooling of the nozzle. Materials of interest are listed in Table 3.4.6-2. Zirconium copper (0.15% Zr) is advantageous in spite of its higher density since it has the highest thermal conductivity properties as well as the highest degree of oxygen compatibility.

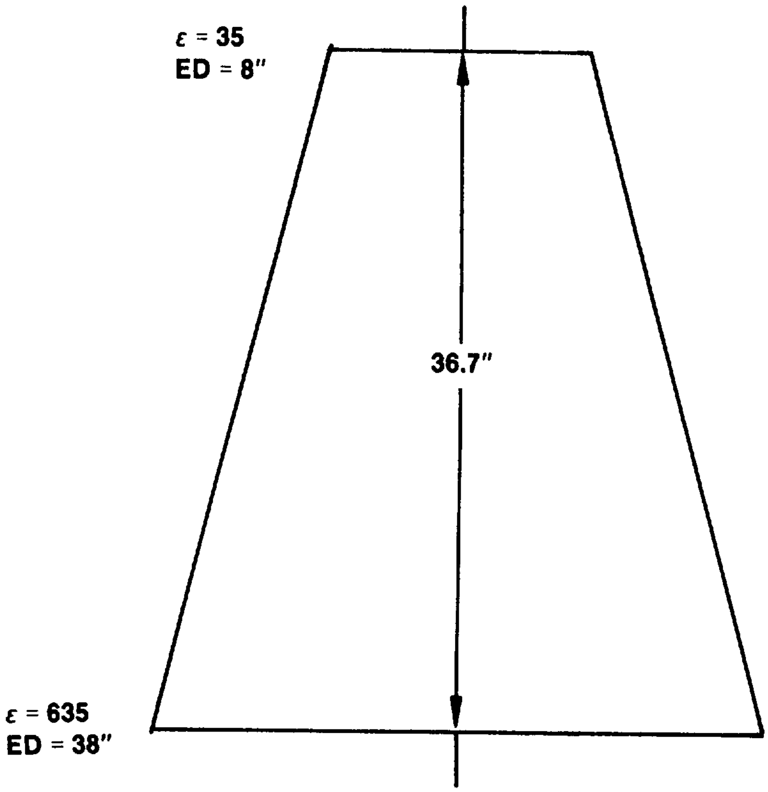
Savings in weight can be achieved either by a low density material or by use of special fabrication techniques. Figure 3.4.6-2 portrays 3 concepts investigated for the 3K OTV TCA nozzle extension concept. Details of these concepts are discussed below.

Use of a brazed tube assembly is a well demonstrated technology. To compensate for the change in diameter from entrance to exit, the tubes can either be swaged to a smaller diameter or else bifurcated.

An alternative method would utilize a slotted cone with an electroformed closeout. This fabrication technique is similar to that used to construct the regenerative cooled thrust chamber.

A third method would use finned tubes. The tubes could be plastically formed from sheet metal with the two halves either brazed or diffusion bonded together. The fin, on either side of the tube, would be cut along a taper so that when the fins overlap for the weld joint, the desired cone angle would be achieved.

The current oxygen cooled nozzle is roughly twice the size of the 3K nozzle extension, however the same fabrication approaches are applicable. A variation of the slotted cone



Total Oxygen Flow:
$\dot{w}_{O_2}$ = 13.39 lb/sec

Figure 3.4.6-1 7.5K Design: Ox Cooled Nozzle Envelope

TABLE 3.4.6-1

OXYGEN COOLED NOZZLE DESIGN PARAMETERS

Ox Flowrate	=	16.2 lbm/sec
Pc	=	2300 psia (overthrust)
MR	=	7
T Ox Inlet	=	640°R
T Ox Outlet	=	860°R
Ox Density	=	15.5 lbm/ft^3
Heat Flux @ e= 28	=	2.7 Btu/sec in.^{2} (maximum)
Maximum Wall Temp	=	1460°R
Maximum ΔP	=	5800 psia
Oxygen Compatible Materials		

Table 3.4.6-2

Oxygen Cooled Nozzle Candidate Materials

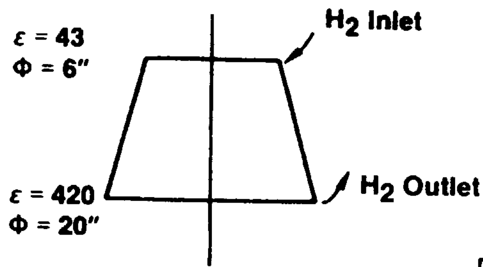
Material Selection

Criteria:

- **Oxygen Compatible**
- **Low Density**
- **High Thermal Conductivity**

Candidate Materials:

	k (Btu/hr ft F)	Density (lbm/in 3)	Y.LS. (ksi)	Ox Rating (Compatibility)
ZrCu	212	0.32	50	1
Nickel 200	44	0.32	14	2
Haynes Alloy #25 or Haynes Alloy #214	13	0.33	35	3
Inconel 600	17	0.30	32	4

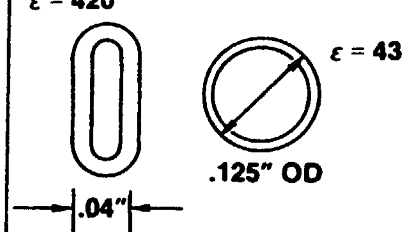
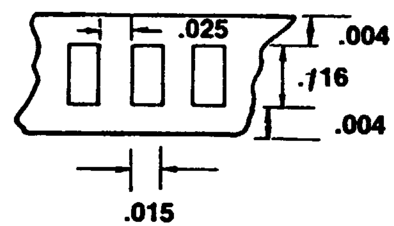


Assumes Split Hydrogen Flow:

$\dot{w}$ = 17% (0.893)

$\dot{w}_{H_2}$ = 0.1518 lb/sec to Nozzle

500 Channels

	Swaged Tube Channels	**Slotted Cone**	**Explosive Formed and Bonded**
	Brazed Tubes (.005" Wall) 	**Slots** 	**Finned Tubes**  0.03" OD
Weight (w/o Manifolds)	5.5#	5.0#	TBD

Swaged Tube Area Ratio:

Inlet/Outlet ≃ 3:1

Figure 3.4.6-2 3.0K Design: Fuel Cooled Nozzle Extension Concept

concept would be achieved by the use of two spun cones. The inner cone, after initial forming would have slots machined onto the outer side for the coolant passages. The outer, thinner cone would provide the close out. A HIP diffusion bond process would join the metals. As part of the HIP process the cones would be bulge or plastically formed onto a mandrel containing the contour of the nozzle. This process is illustrated in Figure 3.4.6-3.

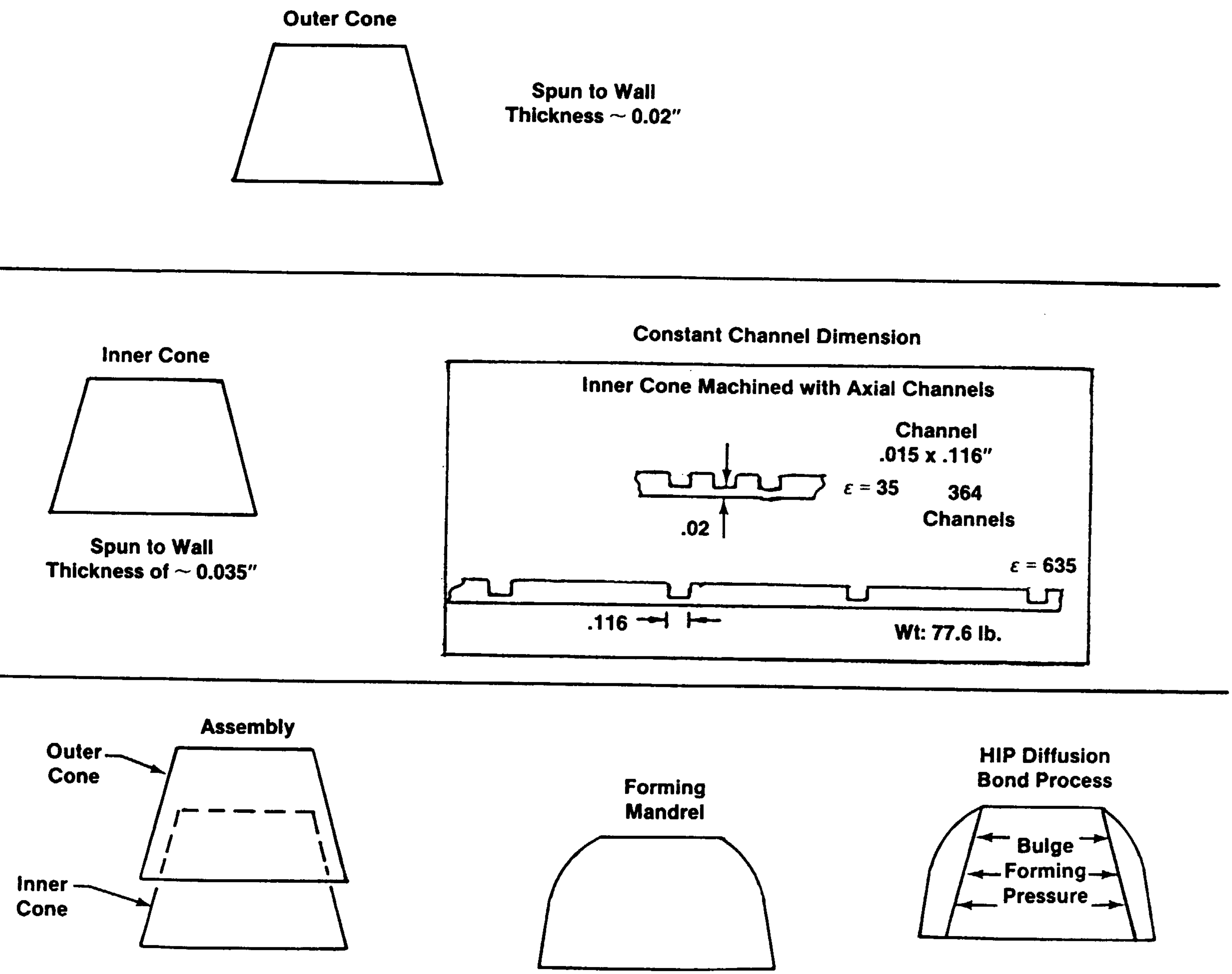
The finned tube approach (Figure 3.4.6-4) identified for the 3K nozzle extension would be directly applicable. A variation of this finned tube concept is illustrated in Figure 3.4.6-5. Half tubes would be welded to the nozzle skirt. The skirt could be formed as a spun cone. Using the HIP process with a mandrel would allow forming the cone into the desired nozzle contour.

Another method to explore for weight savings varies the manifolding. The most straight forward, and lightest method of manifolding would be a single pass system (Figure 3.4.6-6). However, since the oxygen cooled nozzle interfaces with a retractable radiation cooled nozzle, there are space considerations for design of a single pass system. This could be alleviated with the use of a pass-and-a-half system where the exit manifold is moved to the mid-section of nozzle, as illustrated in Figure 3.4.6-7. This would also minimize the number of bifurcations required to go from an entrance diameter of 8 inches to an exit diameter of 38 inches. Figure 3.4.6-7 also shows a concept whereby the tube bifurcation is eliminated by use of a common manifold to redistribute the flow.

A complete thermal analysis still remains to finalize the design of the oxygen cooled nozzle. Parametric studies will assist finalization of the sizing of the coolant passages.

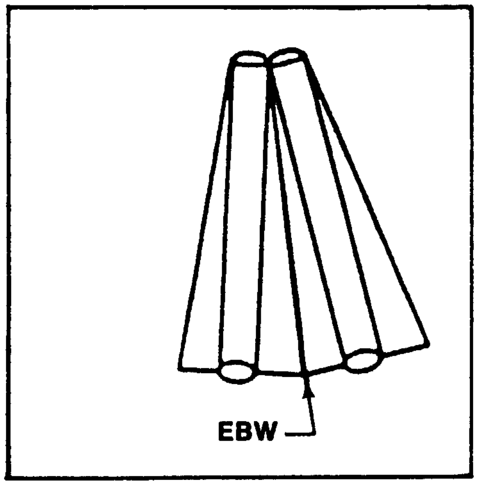
3.4.6.2 Results of Preliminary Thermal Analysis

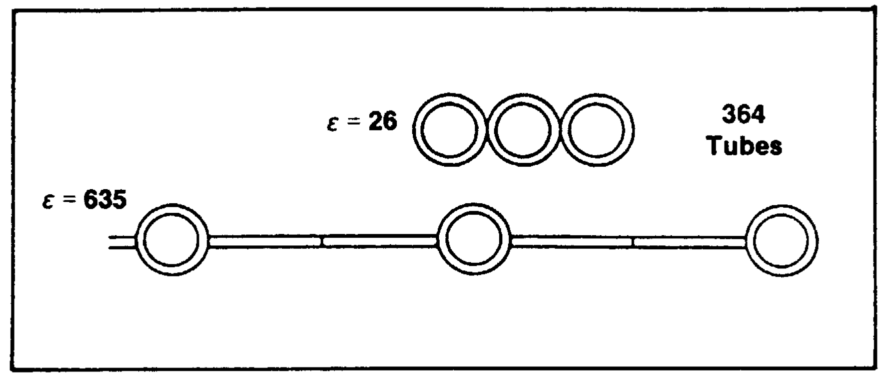
A brazed tube channel assembly was selected for additional analysis, primarily to verify that the oxygen could cool the nozzle adequately. Tube sizes from 1/64 inch to 5/32 inch were evaluated as the basis for the construction. The assumption was that tube bifurcations would be employed as often as needed to maintain the cone shaped structure with swaged tubes. The results of this analysis are given in Table 3.4.6-3. Note that tubes of 1/16 inch diameter and 5/64 inch diameter were near optimum. These are reasonable sizes for a swaged and brazed nozzle construction.



Note: Requires Thermal Analysis to Verify Feasibility

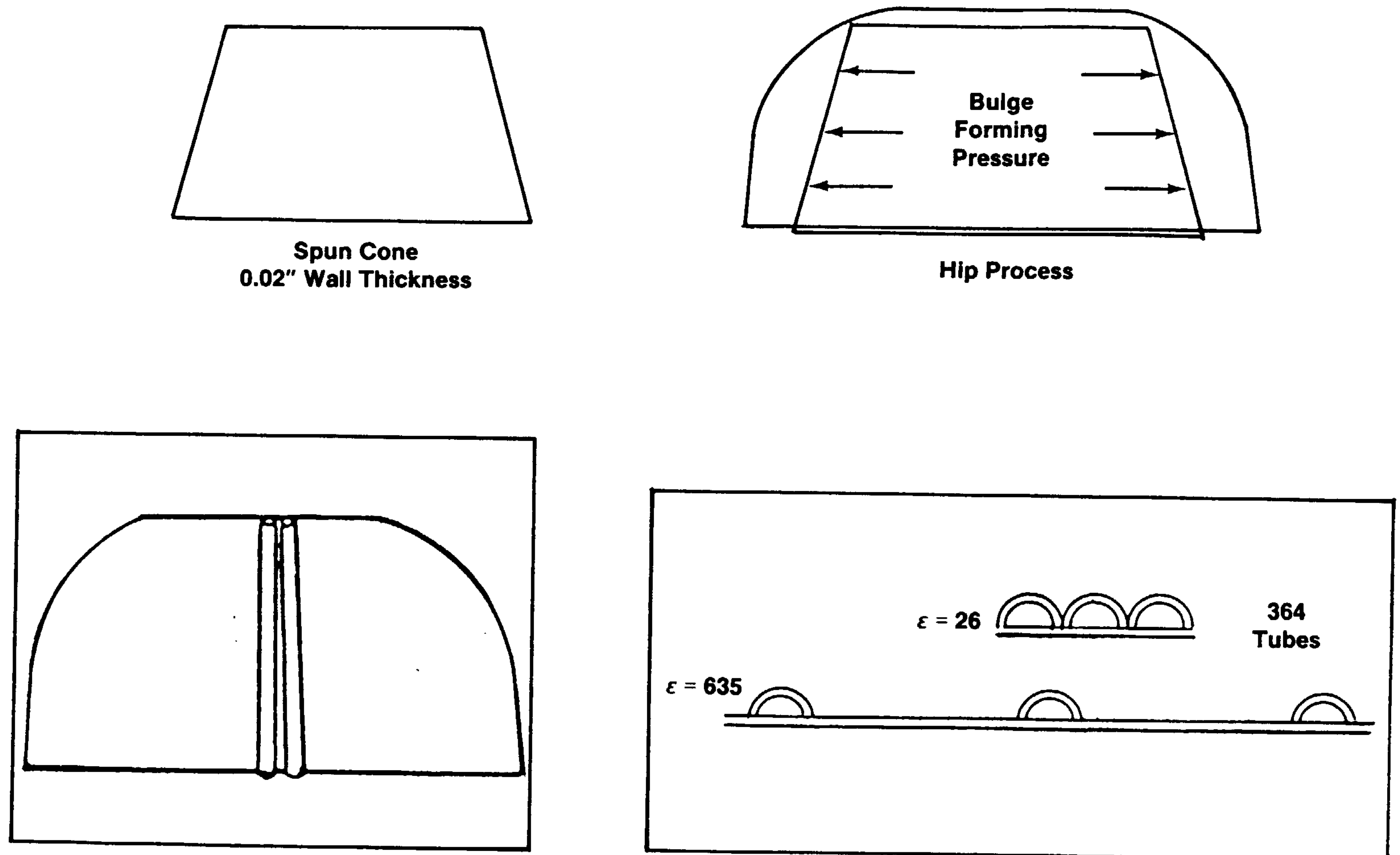
Figure 3.4.6-3 7.5K Design: Ox Cooled Nozzle Concept





Note: Requires Thernal Analysis to Verify Feasibility

Figure 3.4.6-4 75K Design: Ox Cooled Nozzle Finned Tube Concept



Note: Requires Thernal Analysis to Verify Feasibility

Figure 3.4.6-5 7.5K Design: Ox Cooled Nozzle

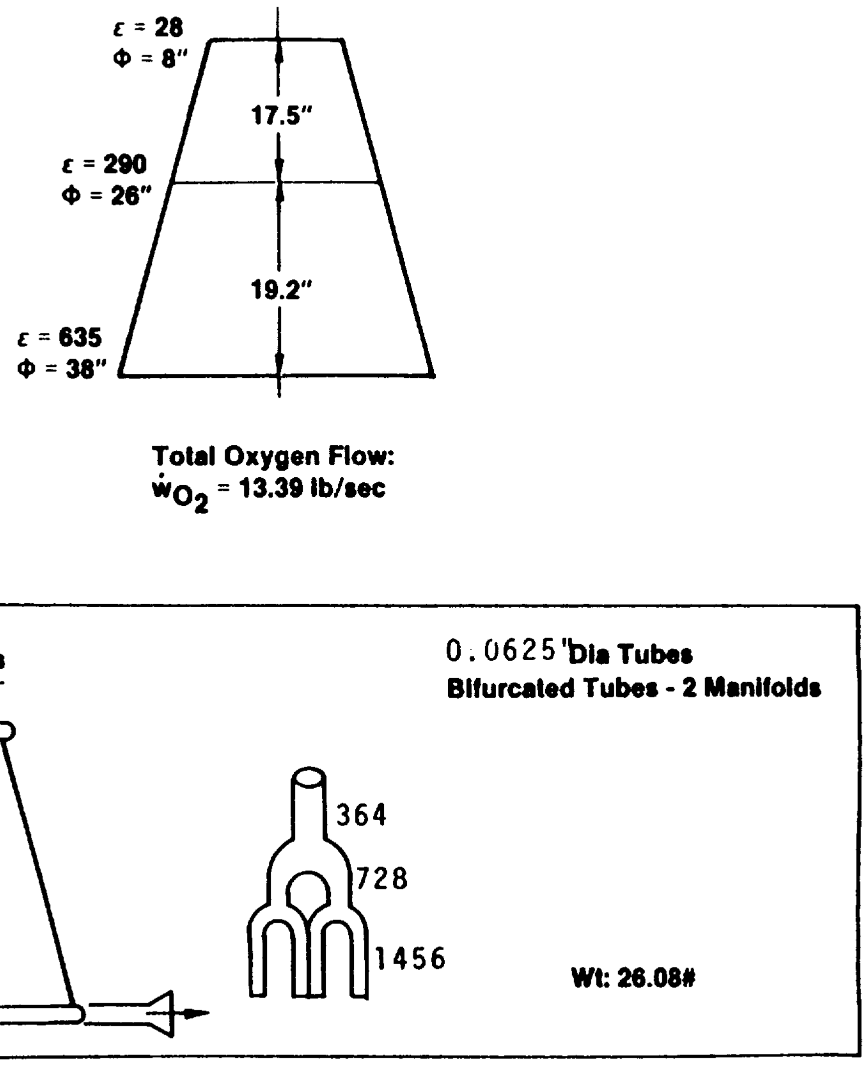


Figure 3.4.6-6 7.5K Design: Ox Cooled Nozzle Extension Concept

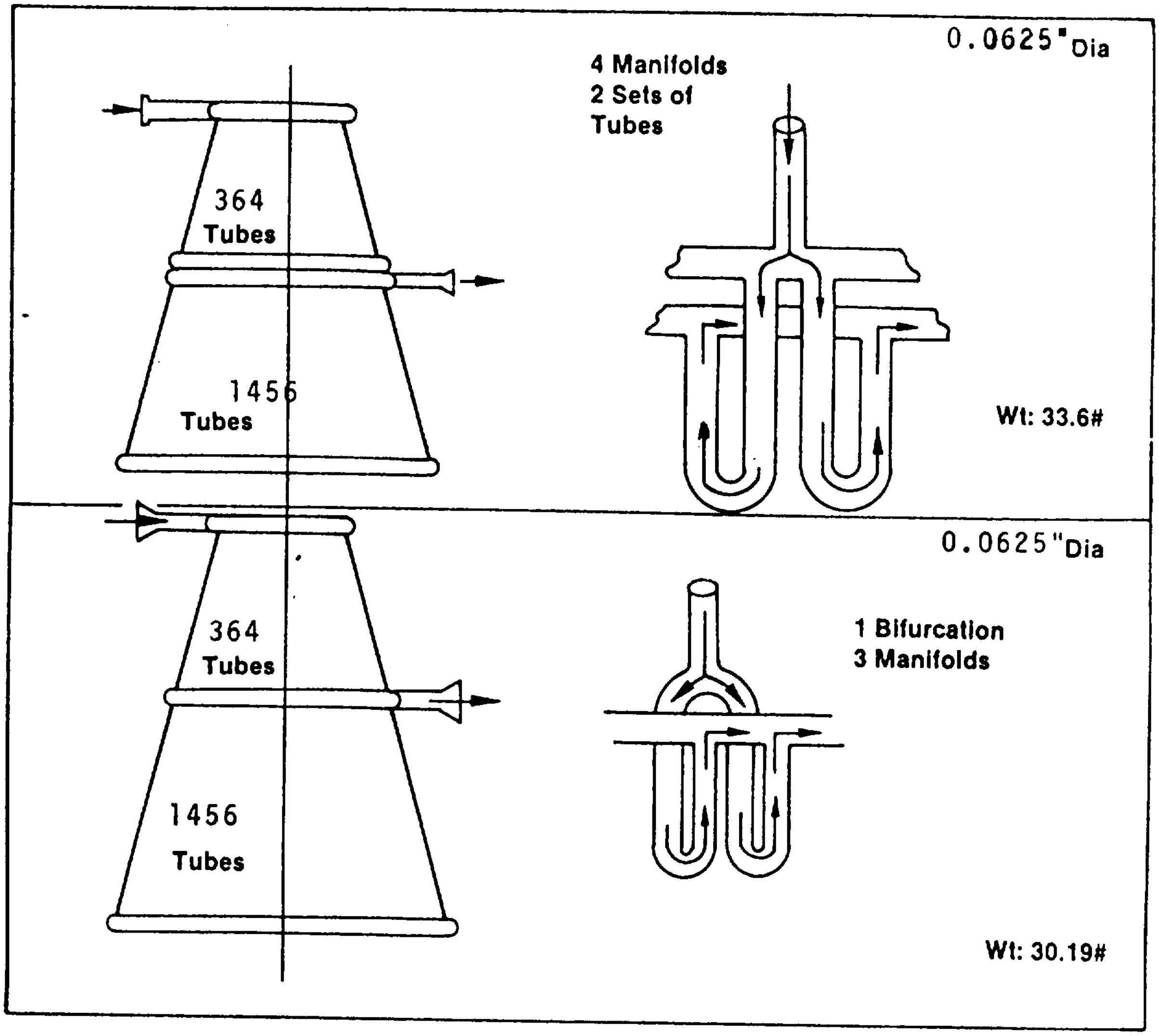


Figure 3.4.6-7 7.5K Design: Ox Cooled Nozzle Extension Concept

Table 3.4.6-3

Design Requirements

Oxygen Cooled Nozzle Coolant Flow & Passage Diameters

Diameter = 8 in. @ Entrance
2t* = 0.007 in.
$\dot{w}$ ox = 13.4 lbm/sec

	Dia (inches)	Area (in²)	No. Tubes	Vel (ft/sec)	Req Vel to Cool	Req 2t
(1/64)	0.0156	0.0002	1113	292	705	0.002
(1/32)	0.0313	0.0008	660	123	196	0.003
(3/64)	0.0469	0.0017	469	77	96	0.005
(1/16)	0.0625	0.0031	364	56	59	0.007
(5/64)	0.0781	0.0048	298	44	41	0.009
(3/32)	0.0938	0.0069	252	36	31	0.010
(7/64)	0.1094	0.0094	218	30	24	0.012
(1/ 8)	0.1250	0.0123	193	26	20	0.014
(9/64)	0.1406	0.0155	173	23	17	0.015
(5/32)	0.1563	0.0192	156	21	14	0.017

* t = Wall Thickness

3.4.6.3 Nozzle Design Conclusions

An optimum nozzle design will require considerable resources and at least a one year design time span. This is a precision component with important thermal requirements and complex hydraulics. It is deserving of development as a separate task.

The optimum material would have high thermal conductivity, good mechanical properties, and oxygen compatibility. Copper alloys do well on conductivity and compatibility, but will require more material because of poor strength. One material considered but doubtful from the oxygen compatibility standpoint is beryllium. If a beryllium alloy could be shown as oxygen compatible it would have to be considered. This is an area where the oxygen/materials compatibility program could be extended to good purpose.

Fabrication techniques are rapidly coming available that could produce a lighter but equally strong cooled nozzle. They should also be investigated in any follow-on development.

3.4.7 Radiation Cooled Nozzle

Design work on this subtask produced far more voluminous results than the assigned hours would indicate. The investigator was both highly efficient and very motivated for the task as it supported other ongoing programs and an expected new NASA MSFC nozzle development program.

3.4.7.1 Ground Rules and Design Issues

This design task was completed under the ground rules listed in Table 3.4.7-1. Of these the most important drivers were the sixty inch envelope constraint and the use of only one extendible/retractable section. The results were developed to address the issues also given in Table 3.4.7-1. Of these the ones given most consideration were weight and fabricability as these greatly influenced the mechanical design.

TABLE 3.4.7-1

OTV ENGINE EXTENDIBLE NOZZLE
GROUND RULES AND ISSUES

Engine Ground Rules:

- Space Based
- 7500-lbF Thrust Nominal
- 115% Thrust Capable
- 6:1 Mixture Ratio
- 2000 psia Chamber Pressure
- 20 Hour/500 Mission Life

Nozzle Ground Rules:

- One Extendible/Retractable Section
- Radiation Cooled
- 60-in. Stowed Length

Extendible Nozzle Issues:

- Weight
- Life & Reliability
- Hot Structural Attachments
- Thermal Isolation
- Maintainability
- Non Destructive Evaluation (NDE)

Deployment System Issues:

- Weight
- Reliability (Fail Safe/Fail Operational)
- Rad-to-Regen Joint Life
- Envelope
- Alignment Tolerance
- Maintainability

The task was started by reviewing the recent literature to determine the state-of-the-art. The most useful reports for this design were:

High area ratio nozzle concepts (R/D, NAS 8-35771)

Evaluation of c/c composites for space engine nozzles Aerojet, NAS 8-35971)

Extendible nozzle tradeoff study, RL10-IIB (P&W, NAS 3-22902)

Titan Stage II extendible nozzle (Aerojet, IR&D)

SRB/SLEEC feasibility study (Aerojet, NAS 8-36571)

3.4.7.2 Concept and Materials Selection

The literature review and Aerojet design experience were combined to produce the design selection summarized in Table 3.4.7-2. The deployment/retraction mechanism is a well proven ball screw system that came through the initial trade study so far ahead of any competitor that it was the only concept carried into the drawing stage of the task. The nozzle material selection was less clearcut. After a review of several candidate materials the choice was reduced to a refractory metal or a carbon composite. The candidate materials are listed in Table 3.4.7-3. Material properties and a recommendation for a gas side wall coating are given in Table 3.4.7-4 for the metals. The material selection is of considerable importance in the design as the weights are different, attachment design is different, and the use of stiffener rings is needed for the metals but not for the carbon composites. The baseline concept design is given in Figure 3.4.7-1. Note that three jack screw rods are baselined. They are driven by three 28 volt DC electric motors which are interconnected by a flexible drive cable system that allows for failure of two of the motors without disabling the system. This is the basic redundancy of the system that adds to the reliability of this basic well-proven device.

3.4.7.3 Nozzle Contour Selection

Aerojet developed a nozzle optimization code that worked in conjunction with the generally available TDK* performance analysis code to analyze the nozzle design for the given envelope and engine parameters. Results of the analysis showed that 100% bell gave the highest specific impulse for a fixed envelope, but 110% bell gave the lowest nozzle weight for a fixed

*Frey, H. and Nickerson, G.R., Two-Dimensional Kinetic Reference Program, prepared by Dynamic Science, Contract NAS 9-10391, December 1970.

TABLE 3.4.7-2

OTV ENGINE EXTENDIBLE NOZZLE
DESIGN CONCEPT SELECTION

Reversible, Ball Screw Deployment
Extension/Retraction System Fixed to Primary Nozzle
Dual Redundant, Flex Shaft Synchronized 28 VDC Motors
Locking/Unlocking Incorporated in Motors
Refractory Metal Finger Seals at Joint
Shoulder Bolt Nozzle Attachment
Columbium and Carbon/Carbon Radiation Section Materials

TABLE 3.4.7-3

OTV ENGINE EXTENDIBLE NOZZLE
RADIATION COOLED MATERIAL OPTIONS

Metals:

- Columbium Alloys C103 and FS-85 w/Silicide or Aluminide Coating
- Nickel Alloys Haynes 230 and Hastalloy X w/NiCrAlY Coating
- Cobalt Alloy Haynes 188 w/CoCrAlY Coating

Composites:

- 2D-Involute Carbon/Carbon (Pyrocarb 409) w/SiC Coating
- Quasi-3D Carbon/Carbon (Novoltex T22) w/SiC Coating
- Quasi-3D Carbon/Silicon Carbide (Sepcarbinox)

TABLE 3.4.7-4

OTV ENGINE EXTENDIBLE NOZZLE
RADIATION COOLED MATERIAL COMPARISON

Metal	Density lb/in.3	Max Temp F	Y.S. @ 2000F KSI	Coating
C103	0.320	2800	18.2	Required
FS-85	--	2800	25.0	Required
Haynes 230	0.319	2100	8.2	Recommended
Hastalloy X	--	2000	6.2	Recommended
Haynes 188	--	2000	-	Recommended

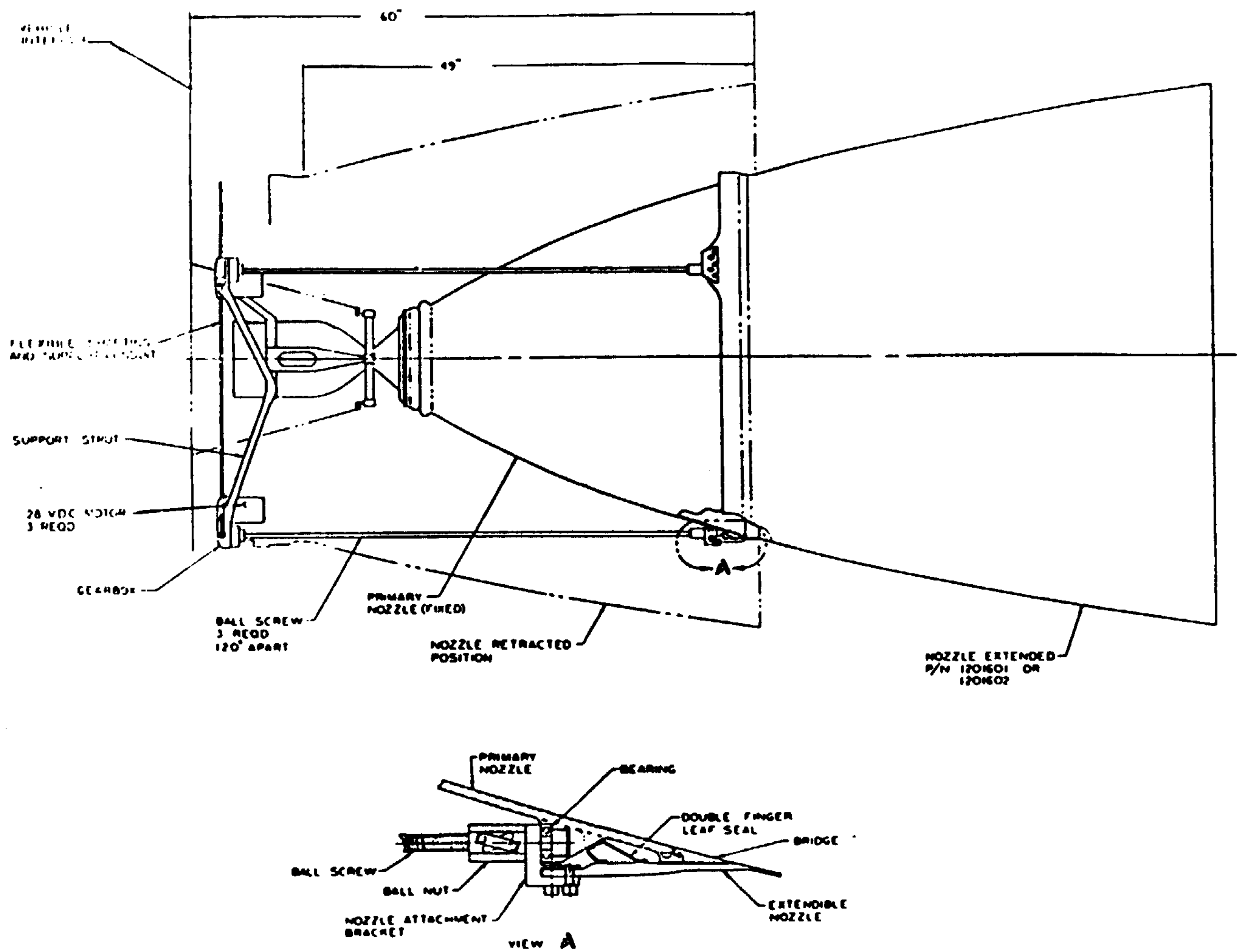


Figure 3.4.7-1 OTV Engine Extendible Nozzle Design Concept

envelope. The selection then had to be made on the payload efficiency. This required use of exchange ratios for the OTV. A search of the Phase A vehicle studies and follow-up calls to NASA MSFC personnel led to use of the following ratios:

$$dW_{pl}/dIsp = 76.3 \text{ lbm/sec}$$
$$dW_{pl}/dW_n = 1.77 \text{ lbm/lbm nozzle}$$

The first relates change in vehicle payload to a change in engine delivered specific impulse (in seconds). The second ratio takes in to account the effect of a larger (heavier) nozzle to improve specific impulse but at the price of a heavier nozzle. For any change of one lbm in nozzle weight the payload weight changes directly by 1.77 lbm.

With the exchange ratios determined the nozzle optimization program yielded the 100% bell as optimum as shown in Figure 3.4.7-2. With the further variable of material density and thickness the optimization in Figure 3.4.7-3 gives a comparison of two columbium nozzles and two carbon-carbon nozzles. The 0.030 in. thick columbium nozzle was selected as the current state-of-the-art baseline. The delta payload to GEO shows that there is a significant payload improvement by going to the lightweight carbon-carbon material.

Presented in Figure 3.4.7-4 is an assessment of the practicality of using more than one nozzle segment. For the columbium materials one segment is optimum. For the carbon-carbon a two segment nozzle would be a performance optimum, but the reliability would be seriously compromised. There was no compelling reason from this analysis to change the one segment nozzle design baseline.

3.4.7.4 Envelope

With the nozzle bell defined and the deployment mechanism defined the nozzle envelope was determined as follows:

Component	Axial Length
Gimbal Clearance (6°)	2 in.
Drive Mechanism/Joint	9
Nozzle Extension	49
Total Envelope	60 in.

All subsequent engine performance estimates are based on the area ratio of 1430:1.

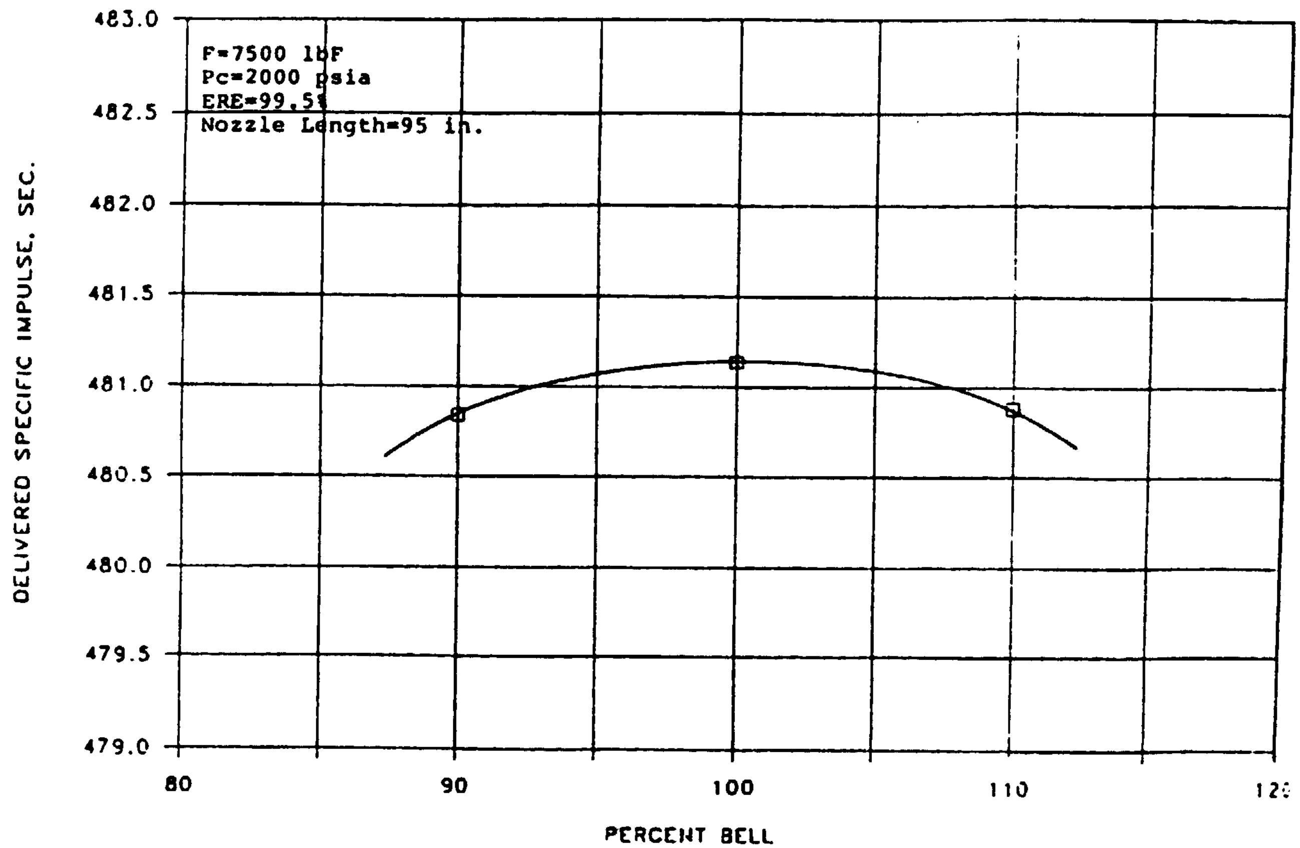


Figure 3.4.7-2 OTV Engine Extendible Nozzle 100% Bell Contour Optimizes Isp

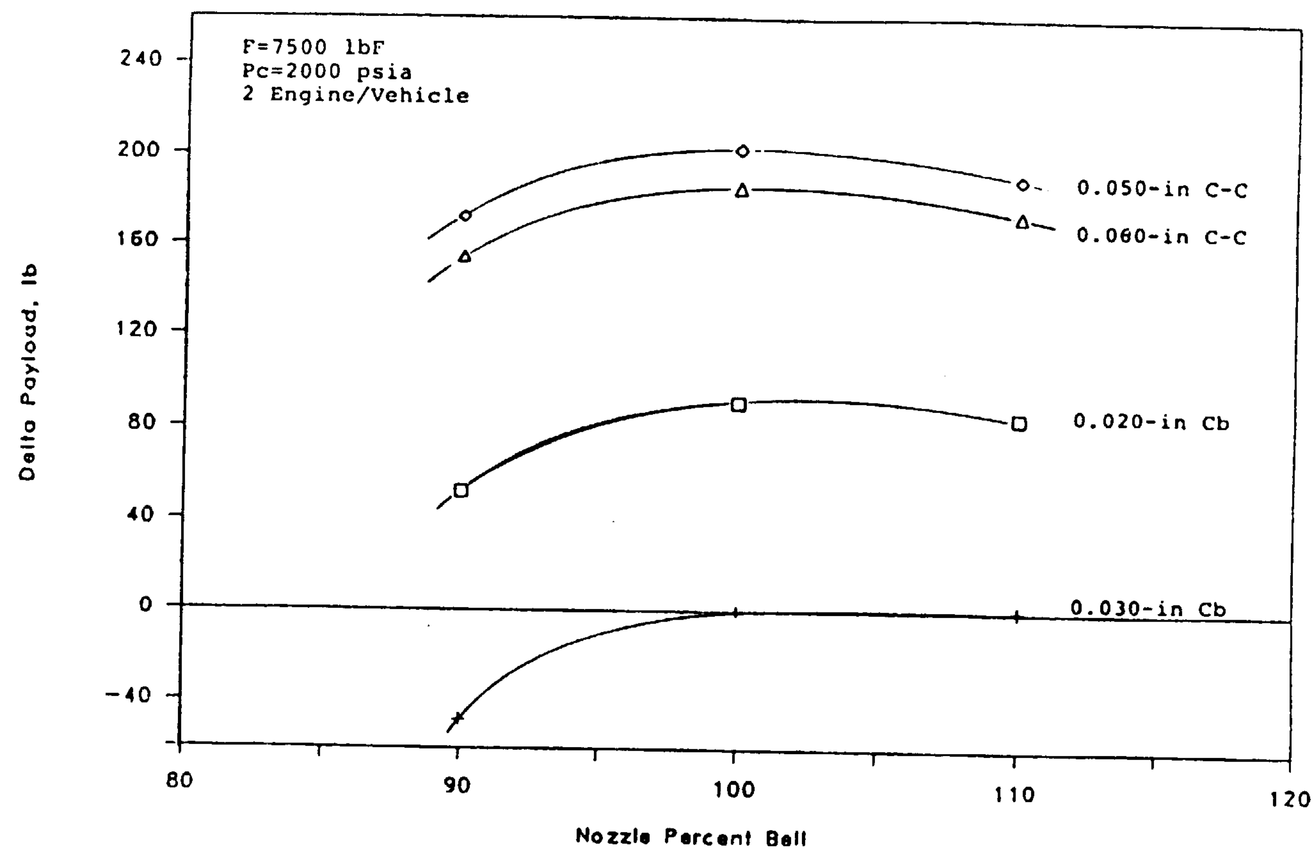


Figure 3.4.7-3 OTV Engine Extendible Nozzle 100% Bell Contour Optimizes Payload to GEO

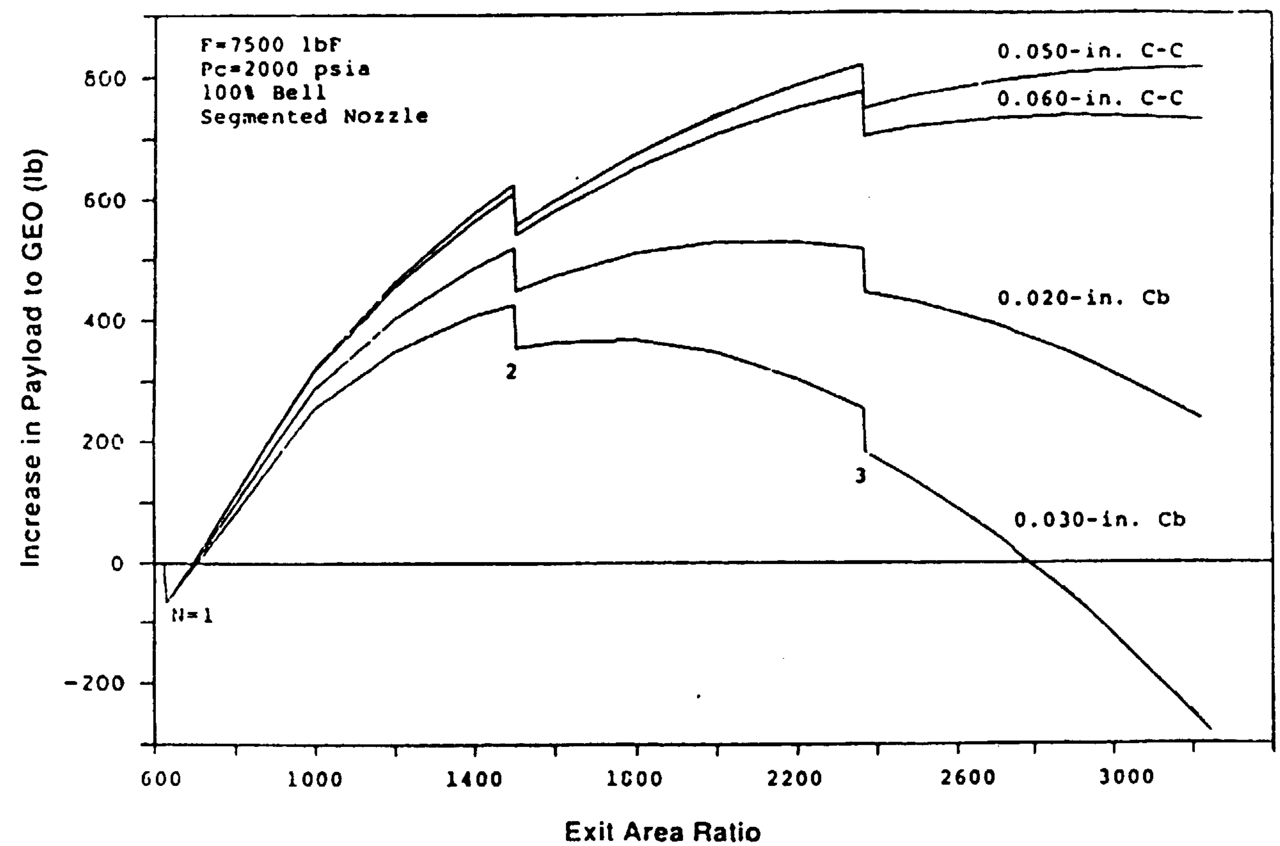


Figure 3.4.7-4 OTV Engine Extendible Nozzle Single Segment Optimum for Metallic Nozzle

3.4.7.5 Thermal Analysis

The OTV engine system assumes two engine (or more) operation with engines side by side. A SINDA (Ref. SINDA 1983/ANSI Code — Systems Imiproved Numerical Differencing Analyzer, prepared by TRW Systems, 1970's) thermal analysis for the configuration is shown in Figure 3.4.7-5. The calculated gas-side heat transfer coefficient versus position starting at the attachment point to the oxygen cooled nozzle is given as Figure 3.4.7-6. With the side-by-side engine mounting two view factors had to be considered. The view factors derived from the SINDA calculation is given as Figure 3.4.7-7. As expected, the inside nozzle section reaches the highest temperatures. Both inside and outside temperatures are plotted in Figure 3.4.7-8. The initially lower temperatures at the lower area ratio positions are explained by conduction cooling from the regeneratively cooled nozzle which attaches to the radiation cooled section with reasonably good conductive surfaces.

3.4.7.6 Mechanical and Thermal Loads

The stress loads for the nozzle were calculated and are presented in Table 3.4.7-5. The concern with this calculation is that OTV operational loads are very poorly defined at this time and the table only presents thrusting loads. The loads from Shuttle deployment and from aerobraking are likely to be the two most severe operations in terms of loads, but they have yet to be defined. A detail design will require knowledge of these loads. Additional calculations were performed to establish stresses for involute carbon-carbon. They are not presented herein due to the restriction previously mentioned, but in all cases the carbon carbon had adequate design margins.

The ball screw sizing was based on a buckling analysis presented in Table 3.4.7-6. The nominal diameter selected was 0.5 inch. A trade between 6061-T6 aluminum, Titanium 6Al-4V, and 4150 steel showed the steel to provide the lightest weight ball screw material.

Also calculated was the thermal strain for various joint materials. This is presented in Table 3.4.7-7. The carbon carbon was the least effected by the thermal environment of the four materials evaluated. A separate calculation was done on the nozzle thermal stress in carbon-carbon versus the initial joint gap. It is not shown due to the restriction, but the stresses for the baseline design are in an acceptable range.

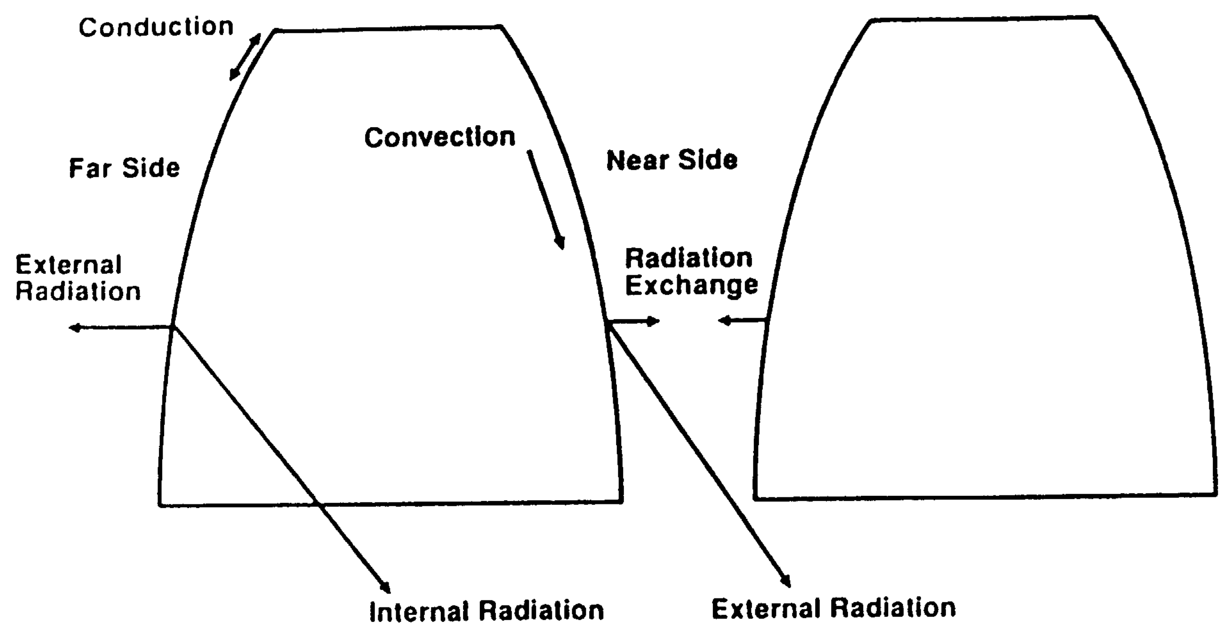


Figure 3.4.7-5 OTV Engine Extendible Nozzle SINDA Thermal Analysis Model

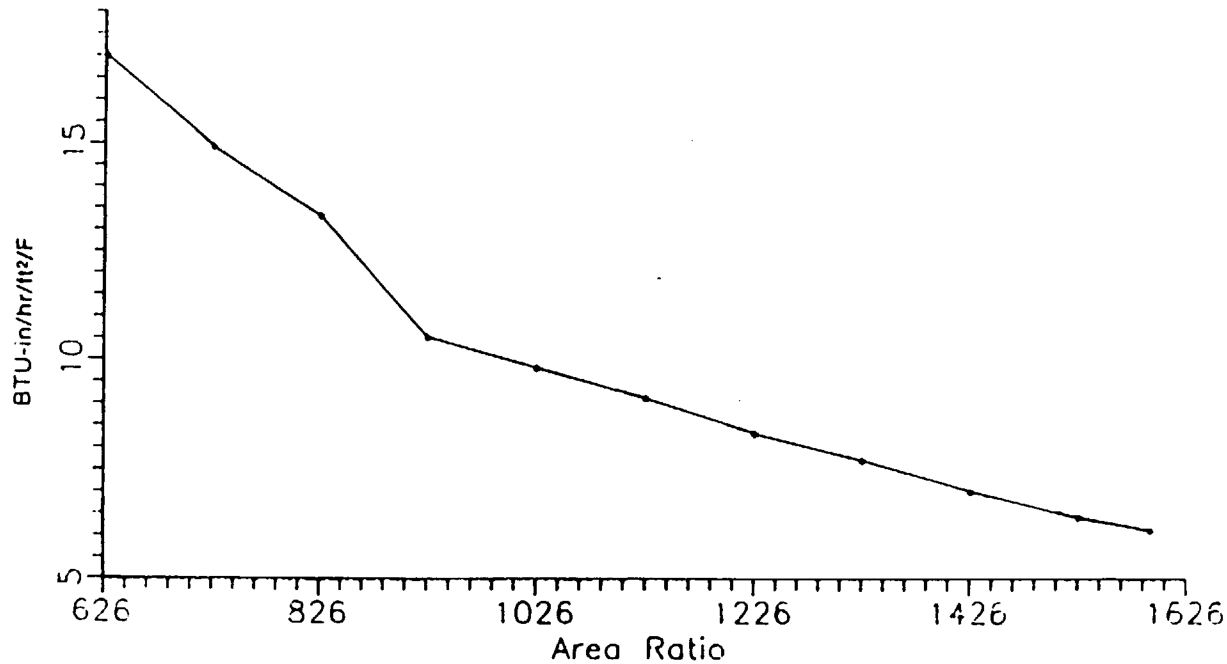


Figure 3.4.7-6 OTV Engine Extendible Nozzle Gas-Side Heat Transfer Coefficient vs. Position

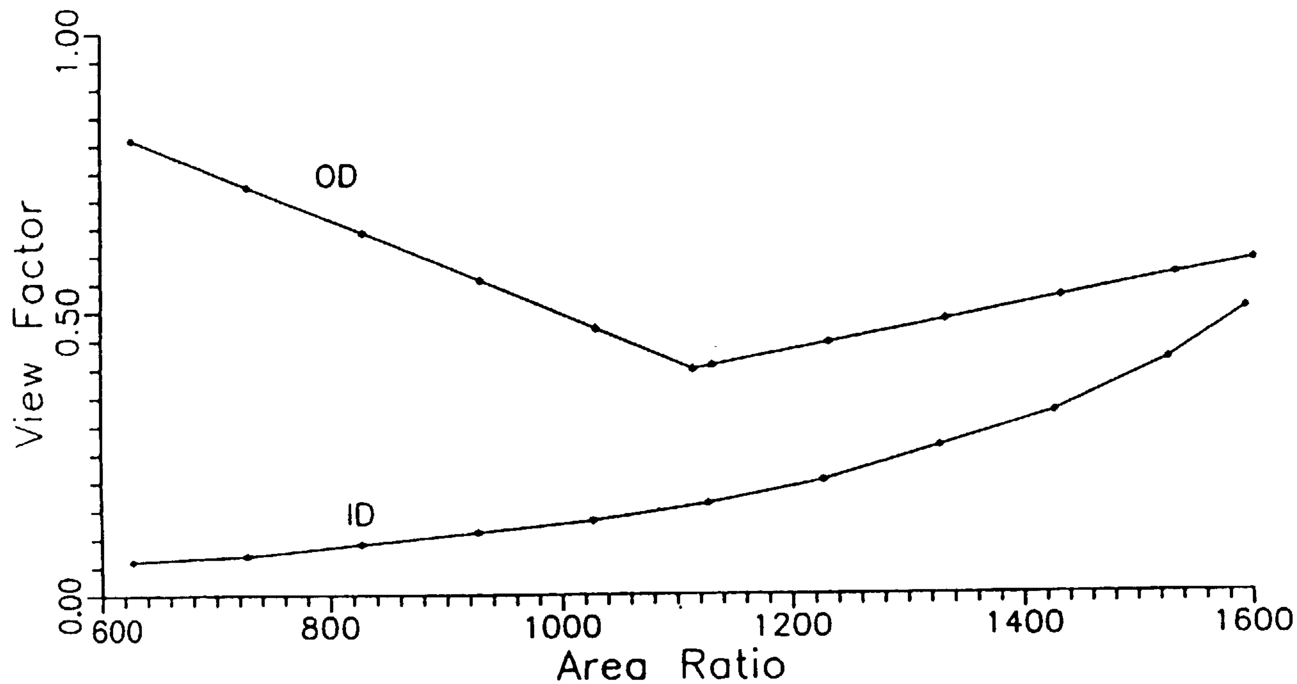


Figure 3.4.7-7 OTV Engine Extendible Nozzle Radiation View Factors vs. Position

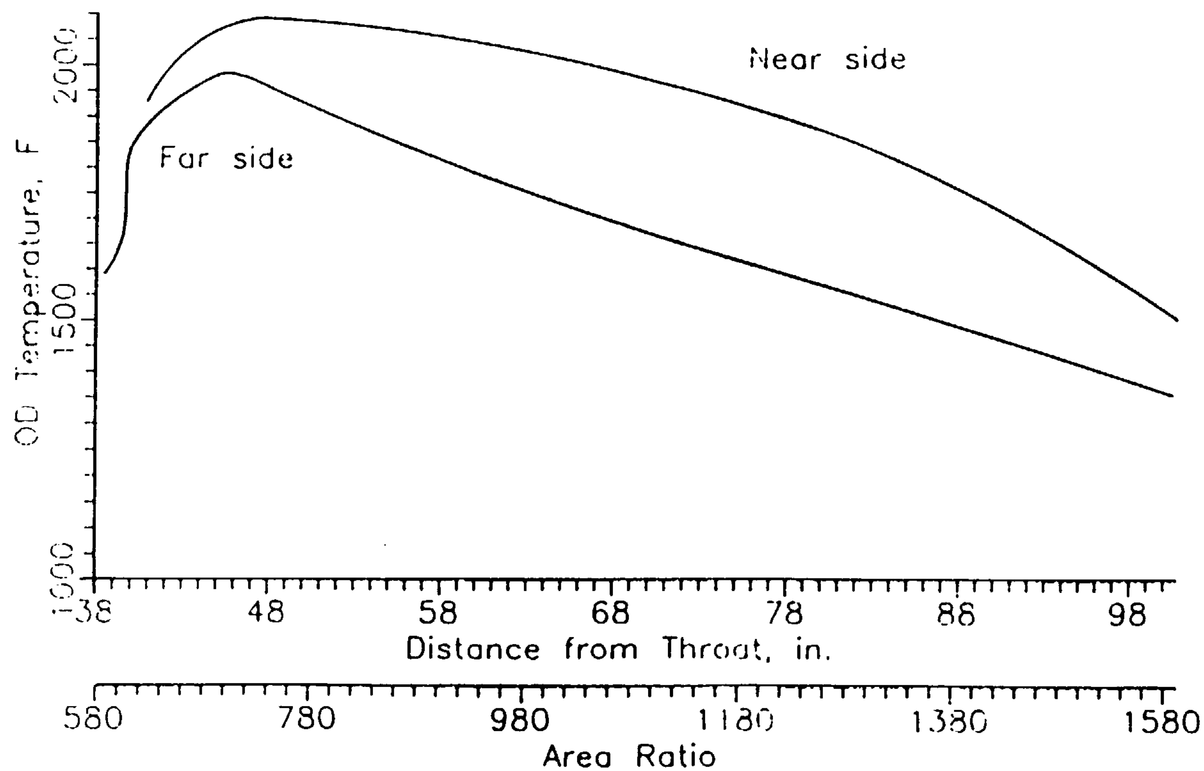


Figure 3.4.7-8 OTV Engine Extendible Nozzle Wall Temperature (C103) vs. Position

Table 3.4.7-5

OTV Engine Extendible Nozzle Thrusting Mechanical Loads are Small

STRESS [1]	*0.020-IN COLUMBIUM* PSI	*0.050-IN CARBON-CARBON* PSI
PRESSURE (HOOP)	128	52
THRUST (AXIAL)	-105	-42
INTERNAL SHEAR (SNUB)	35	6
BENDING (SNUB)	+/-285	+/-56

1. MAXIMUM STRESS CALCULATED FOR THIN WALL REGION NEAR NOZZLE ATTACHMENT

Table 3.4.7-6

OTV Engine Extendible Nozzle Ballscrew Sizing Based on Buckling Analysis

BASIS:

- 100 LBF AXIAL LOAD PER BALLSCREW (3 QTY)
- 55 IN. LONG
- 0.5 MARGIN OF SAFETY MINIMUM (F.S. =1.5)

SELECTED:

- 0.5 IN. NOMINAL DIAMETER
- 4150 STEEL (LIGHTER THAN 6061-T6 OR Ti-6Al-4V)

TABLE 3.4.7-7

OTV ENGINE EXTENDIBLE NOZZLE
THERMAL STRAIN AT REGEN-TO-RAD COOLED JOINT

	α (in./in./°F)	T (°F)	E (in./in.)	Δr (in.)
Beryllium	7.OE-6	440	3.1E-3	0.060
C103	4.2E-6	1550	6.5E-3	0.126
Haynes 230	8.0E-6	1550	12.3E-3	0.240
Carbon-Carbon	1.0E-6	1550	1.5E-3	0.030

3.4.7.7 Gas Side Surface Coatings

The Aerojet analysis showed that both columbium and carbon-carbon materials would benefit from surface coatings for protection from the gas species. The carbon-carbon surface recession after four hours of engine operation is plotted in Figure 3.4.7-9, and for 20 hours in Figure 3.4.7-10. The hump in the curves is due to the cooling effect of the oxygen cooled nozzle near the attachment point. Note that a silicon carbide surface is helpful after four hours of operation but is mandatory for longer service lifes.

Aerojet investigated several coatings for the columbium nozzle. Their effective temperature range and protective life is summarized in Figure 3.4.7-11.

The Aerojet coating recommendations are given in Table 3.4.7-8 along with the baseline materials selections from the metals and the carbon-carbon group.

3.4.7.8 Baseline Nozzle Preliminary Designs

After the trades were completed and the two baseline materials selected the designs were prepared as drawings. The preliminary columbium nozzle design is shown in Figure 3.4.7-12. That for the carbon-carbon is given as Figure 3.4.7-13.

Special attention was paid to the interface and attachment design. In most respects this is the most challenging design problem. Aerojet chose a new concept based on an in-depth review of prior concepts developed at Aerojet, Pratt & Whitney, and Rocketdyne. Of major concern were the issues of:

- Hot gas seal resealability
- Thermal expansion mismatch
- Temperature gradients

The first issue involves the repeated actuation of the nozzle. Prior designs had to deal only with a one time deployment. The Aerojet solution is shown in Figure 3.4.7-14 where the double leaf seal is the basis for the resealable seal.

3.4.7.9 Nozzle Weight

A preliminary nozzle weight was calculated for .020 and .030 inch columbium nozzles without taper and for .050 and .060 inch thick carbon-carbon nozzles without

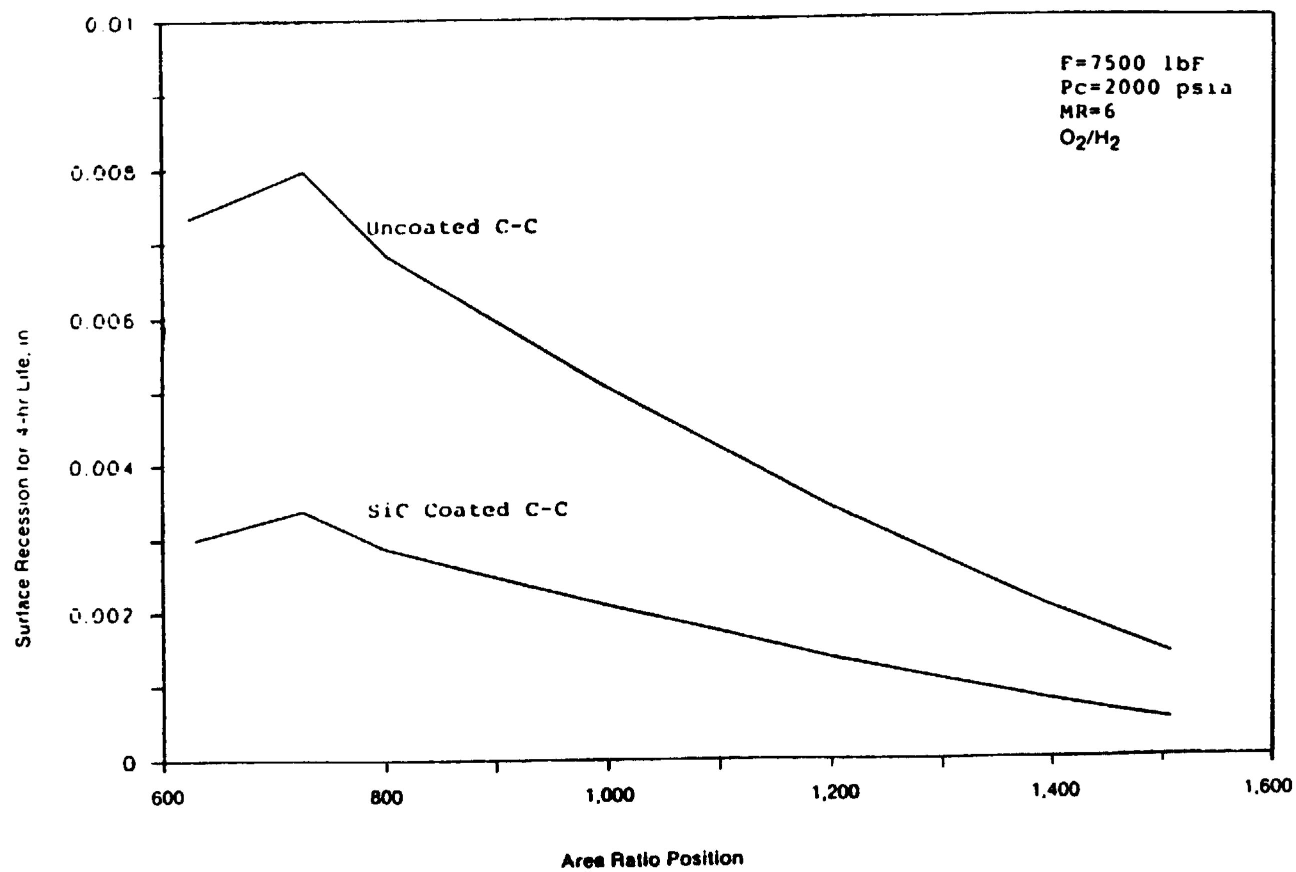


Figure 3.4.7-9 OTV Engine Extendible Nozzle C/C Surface Recession for Service Free Life (4 Hour)

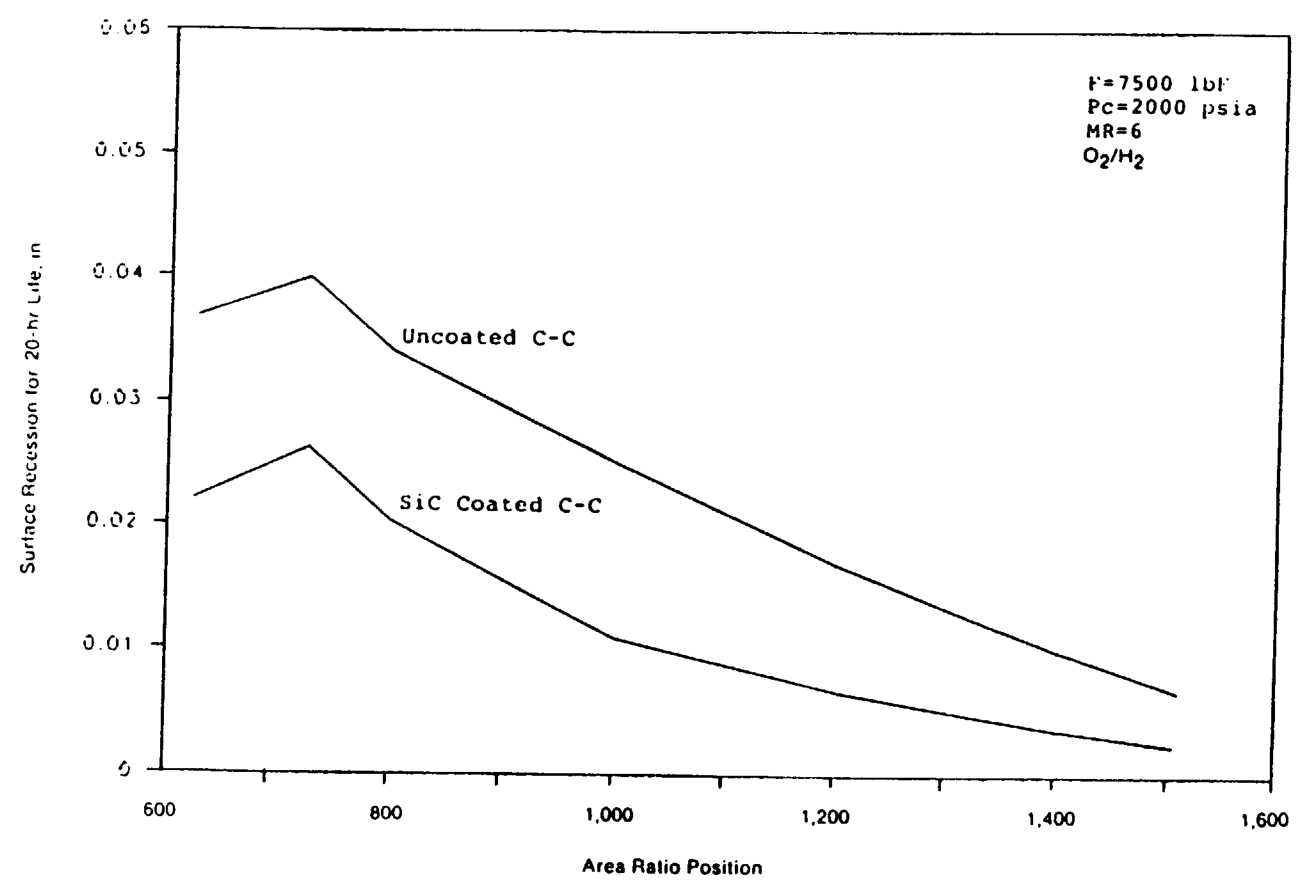


Figure 3.4.7-10 OTV Engine Extendible Nozzle C/C Surface Recession for System Life (20 Hour)

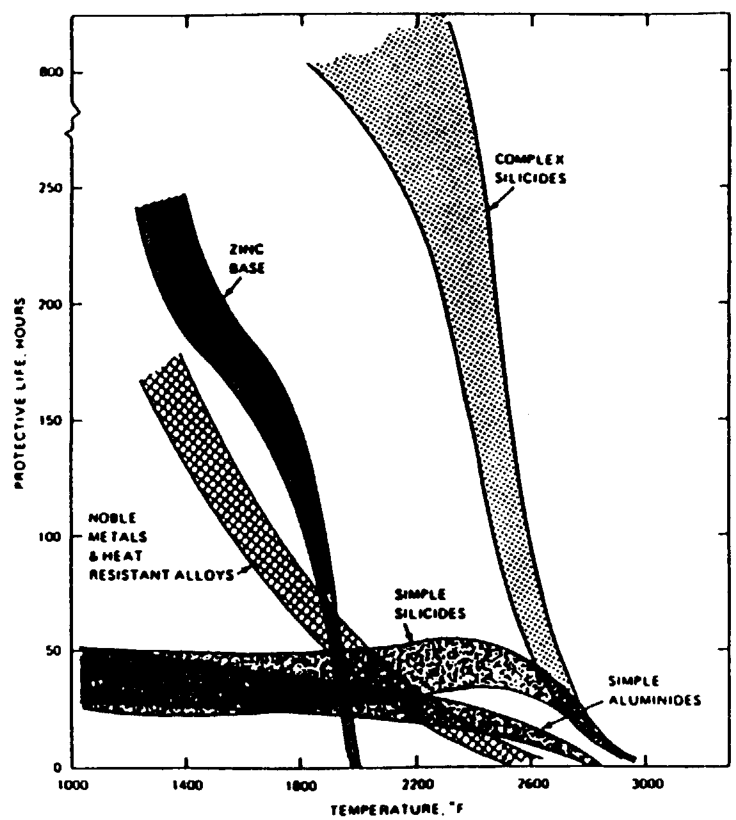
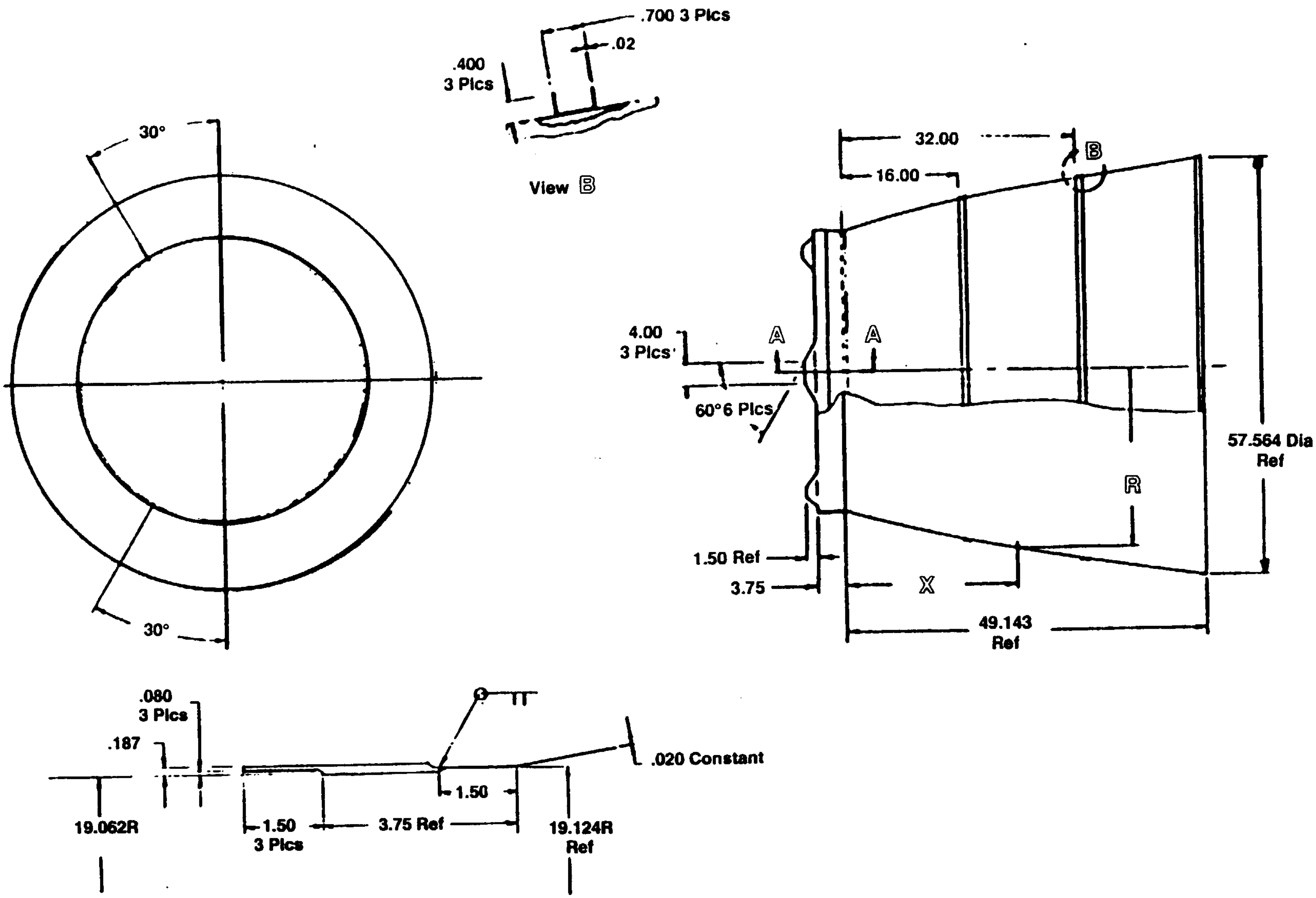


Figure 3.4.7-11 OTV Engine Extendible Nozzle Possible Coating

Table 3.4.7-8

OTV Engine Extendible Nozzle Final Radiation Materials Selection

MATERIAL	*COATING*	*COMMENTS*
C103 COLUMBIUM	FUSED SLURRY SILICIDE (R512)	MATURE MATERIALS & PROCESSES GOOD PROPERTIES HIGH DENSITY
2D-INVOLUTE C/C	OPTIONAL SiC (VLG-25/SiC)	EMERGING MATERIALS & PROCESSES ADEQUATE PROPERTIES VERY LOW DENSITY



Section A – A

Figure 3.4.7-12 Preliminary Columbium Nozzle Design

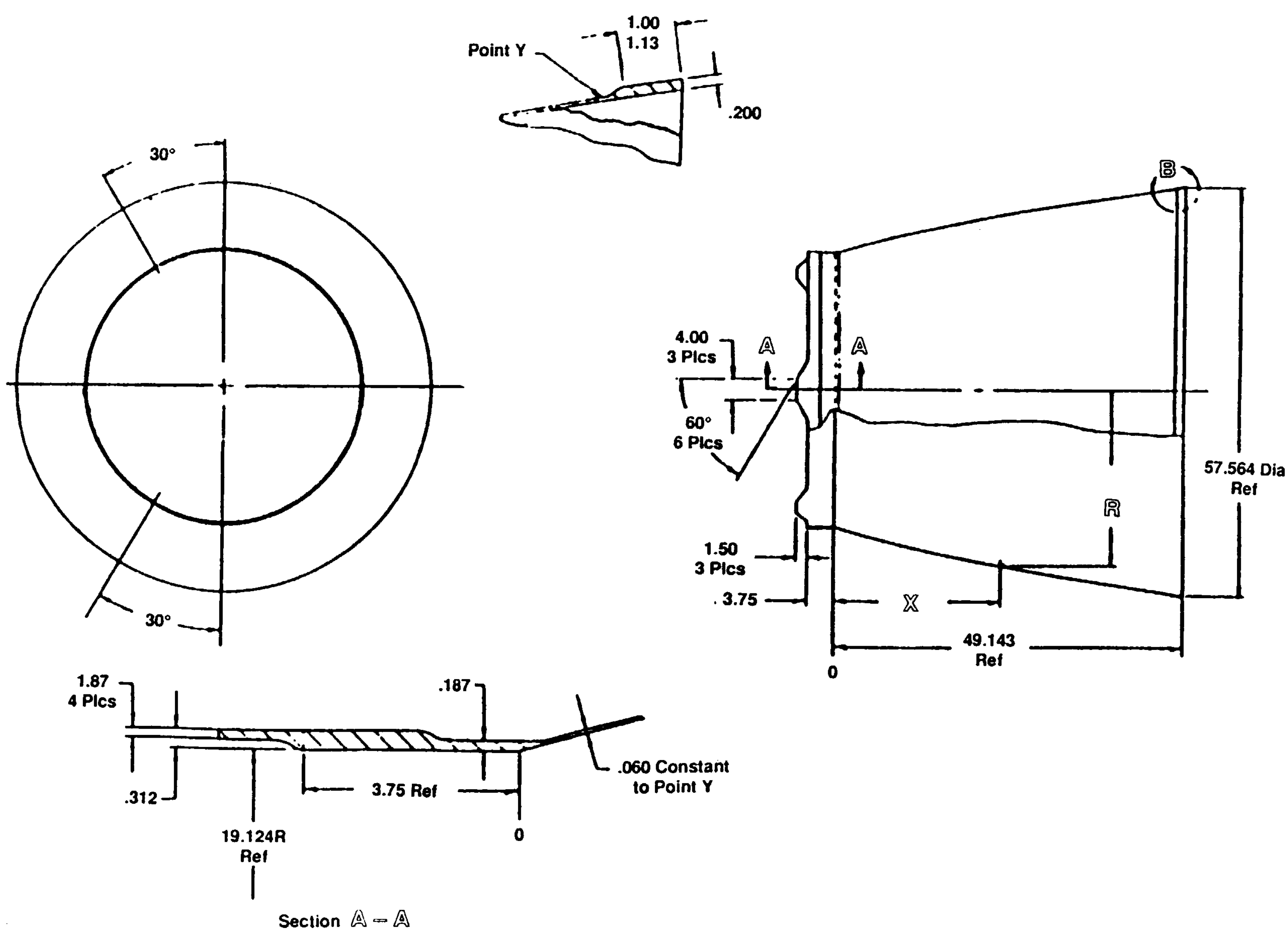


Figure 3.4.7-13 Preliminary Carbon/Carbon Nozzle Design

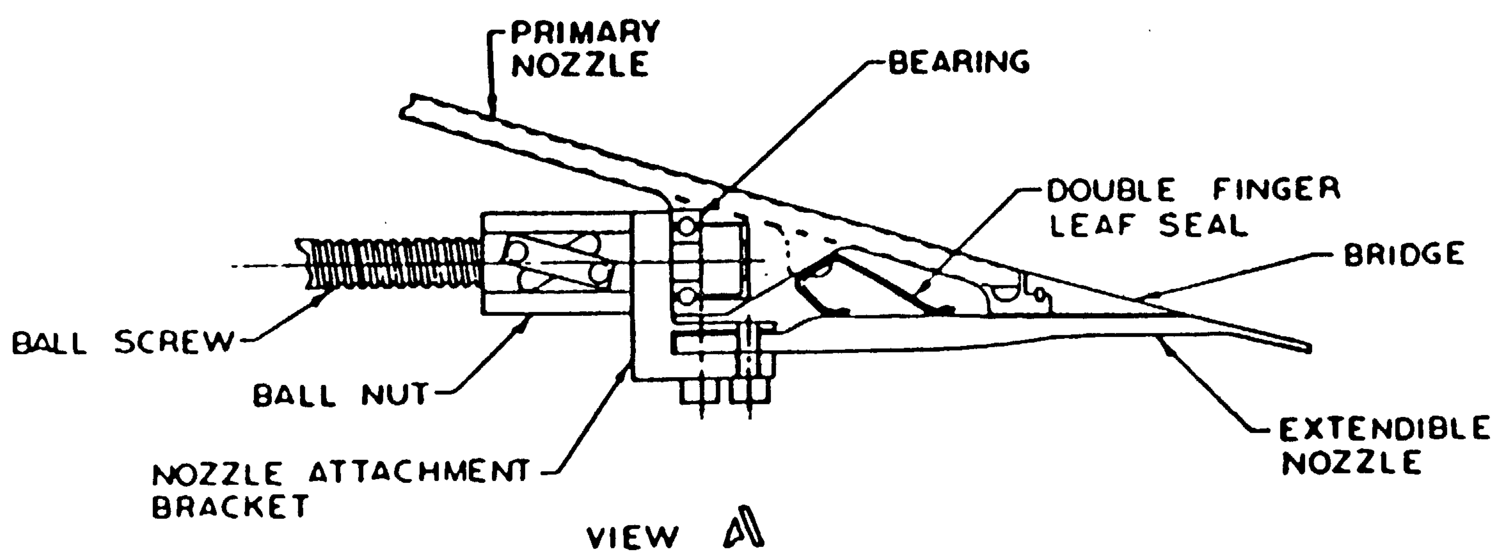


Figure 3.4.7-14 OTV Engine Extendible Nozzle Interface and Attachment Design Concept

taper. In Table 3.4.7-9 the columbium calculations were supplemented with a column for a tapered nozzle thickness. The .060 inch carbon carbon was baselined, its weight summary is given in the third column.

3.4.7.10 Summary

The radiation cooled nozzle subtask was the most thoroughly developed of any for the preliminary design. It is ready for a detailed design and testing program.

Table 3.4.7-9

OTV Engine Extendible Nozzle Preliminary System Weight Estimates

	C103 0.028/0.018/0.010 IN.	C103 0.020 IN.	2D C/C 0.060 IN.
NOZZLE SKIN	35.4	48.3	24.7
NOZZLE ATTACHMENT	18.1	18.1	7.0
NOZZLE STIFFENER	4.7	4.7	1.5
BALLSCREW (3)	8.4	8.4	8.4
GEARBOX (3)	8.1	8.1	8.1
BALL NUT (3)	5.4	5.4	5.4
28V DC MOTOR (3)	9.0	9.0	9.0
FLEX CABLE (6)	4.2	4.2	4.2
SUPPORT STRUTS	5.0	5.0	5.0
TOTAL (PER TCA)	98.3	111.2	73.3

3.5 THRUST CHAMBER DESIGN

3.5.1 Background/Introduction

The thrust chamber design documented in this section was developed under Task C.3, 7.5K Thrust Chamber Assembly (TCA) Final Design.

OTV engine studies have identified a dual propellant expander cycle as offering advantages over either staged combustion or gas generator cycles. The dual expander cycle requires no interpropellant seals for the turbopumps. In the dual propellant expander cycle, the fuel is heated to drive the fuel turbopump and the oxidizer is heated to drive the oxidizer turbopump. The two pumps are totally separate eliminating the design problems and life limitations associated with interpropellant seals on the turbomachinery.

Operation at a higher chamber pressure (Pc) is also possible by the increased amount of thermal energy available to drive the turbopumps. The dual propellant expander cycle is fairly simple, plumbing is straightforward, and it offers excellent performance potential. Gasification of both propellants provides a convenient source of tank pressurant and allows gas-gas injector elements to be used. Gas-gas injection is desirable in that it enables full range throttling with a single injector, without significant performance or stability degradation. Both propellants are routed to the injector for combustion after driving the TPA turbines.

Documented in this report is the evolution of the TCA design for a dual propellant expander cycle at an uprated thrust level of 7500 lbf with engine performance, weight, and reliability requirements considered. Early in the OTV program, a design for a 3000 lbf thrust level TCA was developed. The TCA design (Figure 3.5.1-1) consisted of an annular injector with an oxygen cooled centerbody housing the oxygen turbopump. This concept was evaluated using data from hot fire tests with calorimetric TCA hardware. Information obtained from these 3.0K lbf thrust hot fire tests was used in the thermal design of the 7500 lbf thrust TCA.

The approach to the upscaled 7500 lbf thrust level engine was to use the 3K lbf thrust level design concepts and incorporate modifications necessary to meet the new operational requirements and state-of-the-art fabrication criteria. The basic dual propellant expander cycle has been retained, and a new TCA design conceptualized in an effort to minimize weight, maximize performance, and improve reliability.

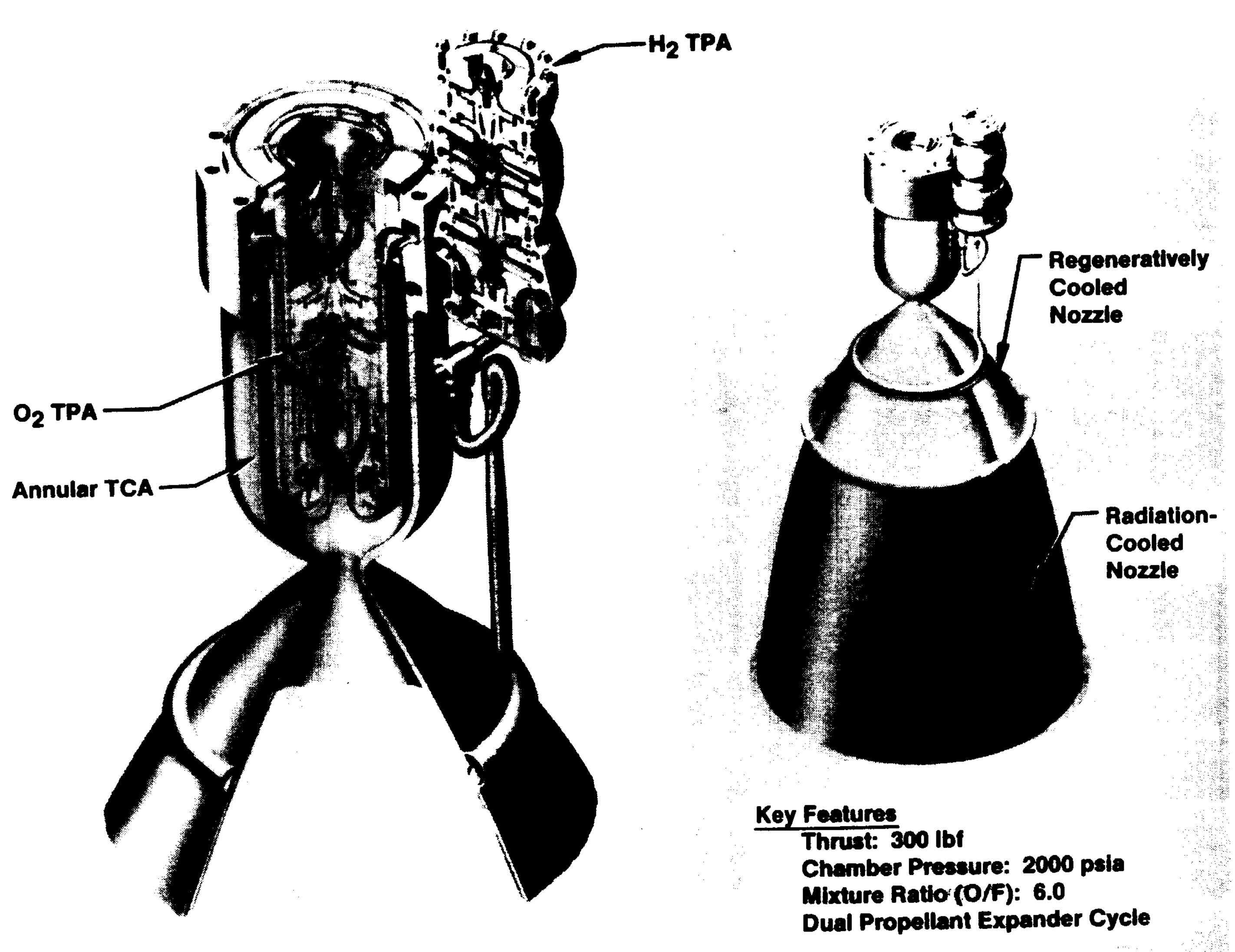


Figure 3.5.1-1 ATC 3K lbf Thrust Level Engine for OTV Propulsion

The design of the 7.5 K lbF thrust level TCA for the Orbital Transfer Vehicle Engine (OTV-E) was based on the system parameters in Table 3.5.1-1.

3.5 SUMMARY

The baseline chamber for this program was a single pass regenerative design to be made from NASA-Z Copper with conventionally milled coolant channels. A high strength electroformed NiCo alloy would form the channel backside closeout. The nominal L' (injector face to throat length) is 9.8 inches with a contraction ratio (Cr) of 17:1." An oxygen cooled tube bundle nozzle extension is attached at an area ratio of 28:1. This concept is shown in Figure 3.5.2-1.

NASA-Z was considered to be the chamber material of choice due to the higher strength at elevated temperatures and high thermal conductivity properties. The upper gas side wall temperature limit used in the heat transfer analysis was 1460 degrees R. Electroformed NiCo was selected for the channel closeout material based on the increase in strength (Figure 3.5.2-2) over that of EF Nickel. With the increased strength of the selected materials, a decrease in weight is obtained by the reduction of the closeout wall and an increase in cycle life is obtained by the decreased thermal strain resulting from a decrease in the temperature gradient.

Sizing of the coolant channels is based on demonstrated fabrication capability with copper chamber liners. The minimum channel and land widths have been demonstrated to be 0.01 inches with aspect ratios (depth to width) of the coolant channel to be a maximum of 10:1.

The number of coolant channels in the liner is determined by the throat diameter. A smooth transition in channel width and depth is desirable as the channel geometry varies from the barrel section through the throat section. Straddle milling the channel width profile is possible but difficult to do. Step machining is possible also, but the pressure drop penalty is high. Bifurcation of the channels would provide another method for transitioning the channels through these regions of varying diameters.

The performance of the dual propellant expander cycle is dictated by the power balance requirement. This establishes the entrance and exit pressures and temperatures of the thrust chamber and injector at the various operating conditions. Because the OTV engine is designed to be throttleable through a 10:1 range, the system power balance has to be verified at

TABLE 3.5.1-1

TCA DESIGN PARAMETERS

Propellants	LH_2/LO_2
Performance	480 lbf-sec/lbm, @ e = 1430:1
Weight	360 lbm
Length	120 inch max, 60 inch stowed length
Life	20 hours, 500 restarts
Reliability	Man Rating
Maintainability	Space Based
Nominal Operation	MR = 6.0 Pc = 2000 psia LO_2/LH_2 Thrust = 7500 lbf Throttling range 10:1

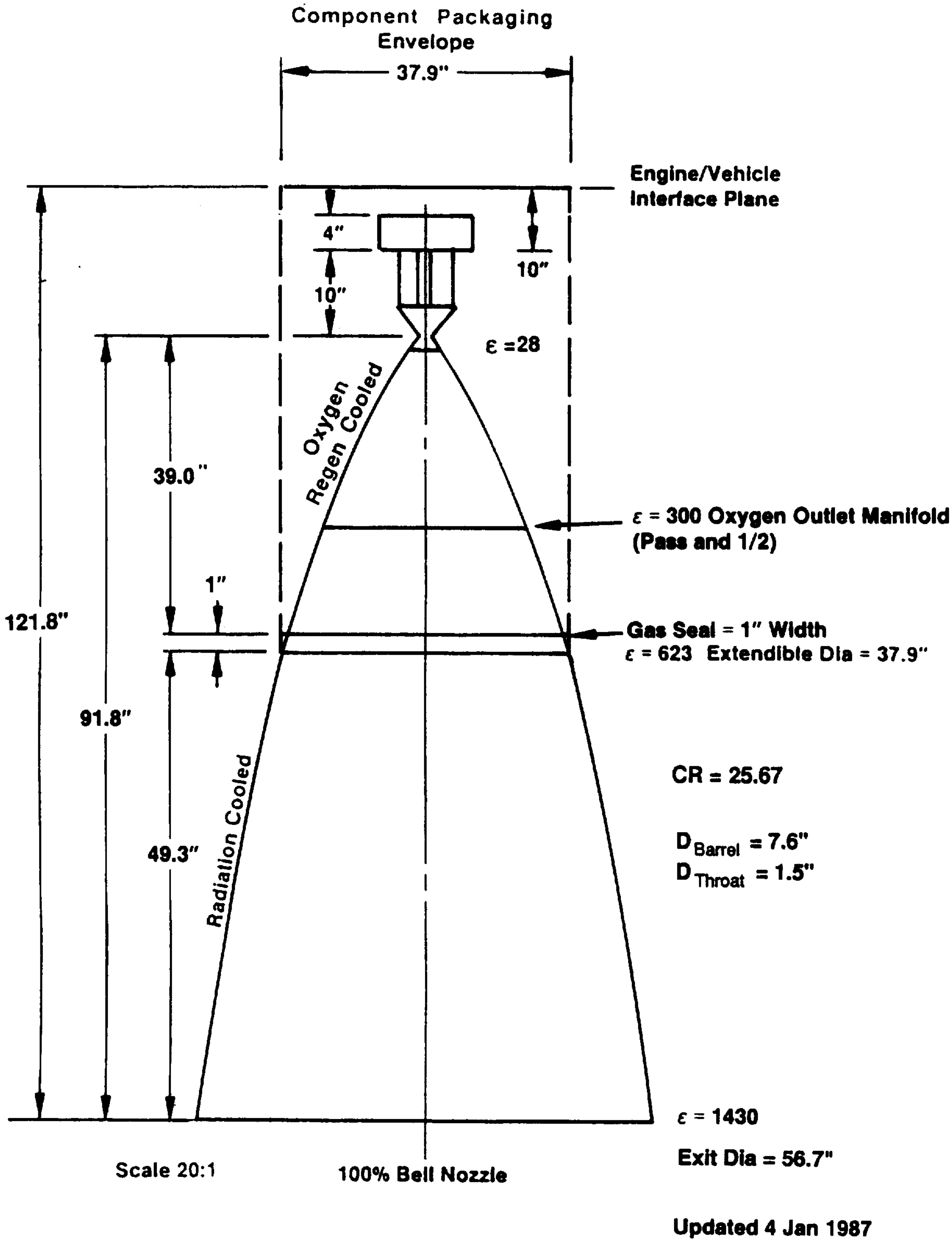


Figure 3.5.2-1 OTV 7.5K lbf Engine – Major Dimensions

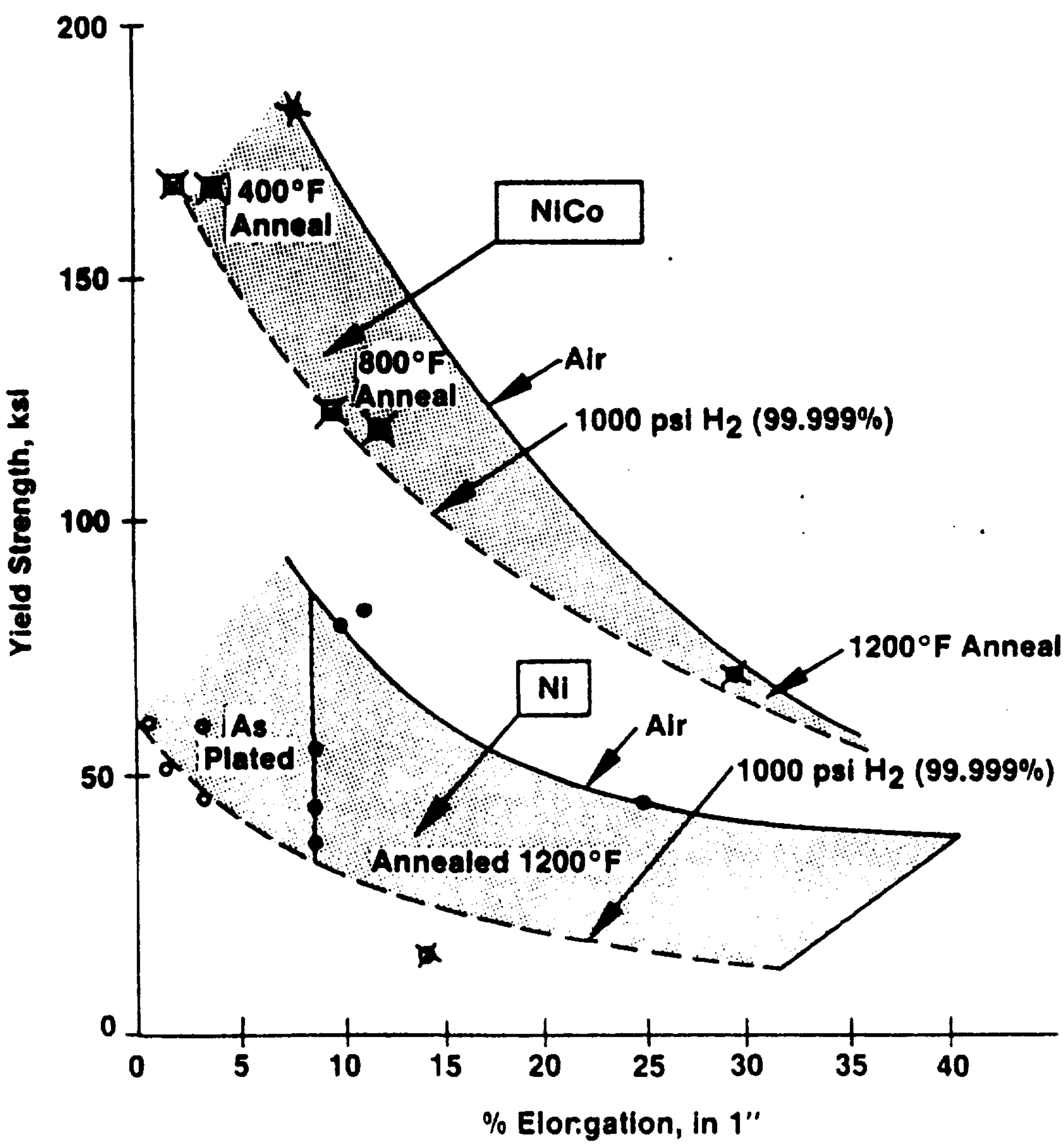


Figure 3.5.2-2 Properties of Electroformed Nickel and Nickel Alloys

each corner of an operating map as shown in Figure 3.5.2-3. This enabled calculation of the required thrust chamber surface area to ensure proper transfer of heat to the TCA components.

The heat input to and the pressure drop of the coolant through the regen jacket was estimated by the SCALE computer program (Reference 2). The information is used to provide the proper pressures and temperatures of the working fluid at the turbine inlet for the power balance calculation. The SCALE program is used to determine the channel depth for an input channel layout (channel width and land width) required to satisfy limitations on gas side wall temperature (material limit), aspect ratio (fabrication limit), and coolant channel gas-side to back-side wall temperature difference (strain and cycle life limit). The pressure drop and bulk temperature rise of the coolant is calculated based on the design geometry.

Methods have been explored to enhance heat flow to the propellants including: (1) increasing the hot gas-side wall area by roughening the surface; and (2) increasing the free stream velocity to increase the heat flux. Earlier studies emphasized designs providing additional surface area exposed to the combustion gases. Later studies aimed at meeting the high cycle life (>500 cycles) found that a "mini-channel" design reduced the temperature differential induced strain thereby increasing cycle life. The mini-channel concept shown in Figure 3.5.2-4, has been retained for the 7.5K lbf thrust level OTV TCA design.

3.5.3 Baseline Design

The initial 3K lbf thrust level TCA design included a hydrogen cooled outer chamber and throat, an annular 2-row injector, and an inner body with oxygen regenerative cooling. A split flow scheme was devised whereby 83% of the hydrogen flow was used to cool the regenerative chamber. The remaining 17% was routed to cool the aft nozzle from an exit area ratio of 43:1 to 420:1. Schematically, the hydrogen coolant was split and then flowed in parallel to minimize the pressure drop in the overall system. In scaling up the TCA from a 3K lbF thrust level to a 7.5K lbF thrust level, the guidelines listed in Table 3.5.3-1 were used as a design "kick-off" point.

The 7.5K lbf TCA design was started with a review of the 3.0K lbf TCA concept. After reviewing the initial concept on the basis of maintainability and reliability, the oxygen turbopump assembly was removed from the middle of the combustion zone. A primary advantage of placing the oxygen turbopump in the center of the combustion chamber had been the simplification and reduction in plumbing necessary to feed the oxygen pump and turbine. The concept had utilized oxygen cooling of the cylindrical turbopump housing (centerbody) to derive the energy for

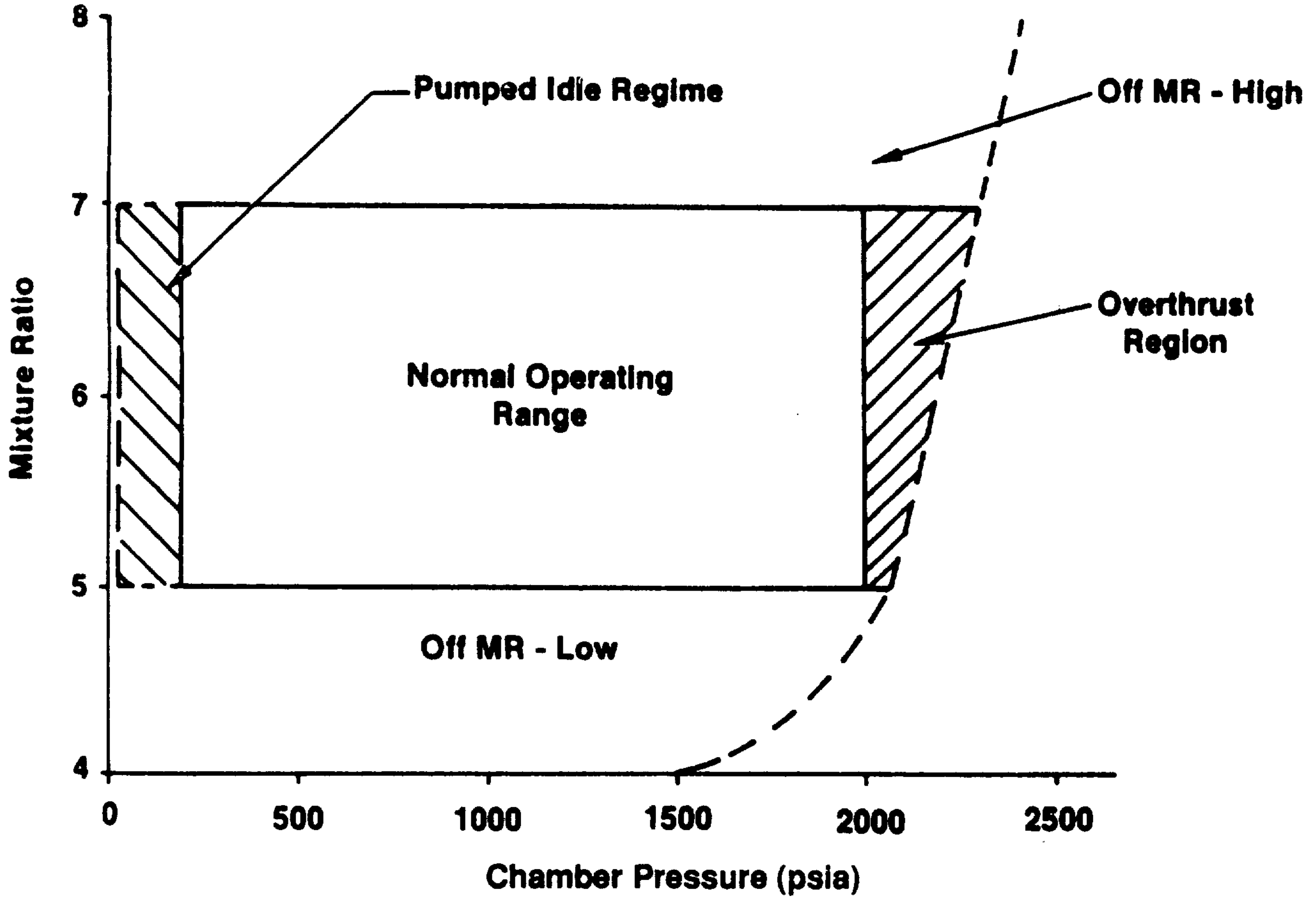


Figure 3.5.2-3 7.5K lbf Thrust Level OTV Engine Operating Envelope

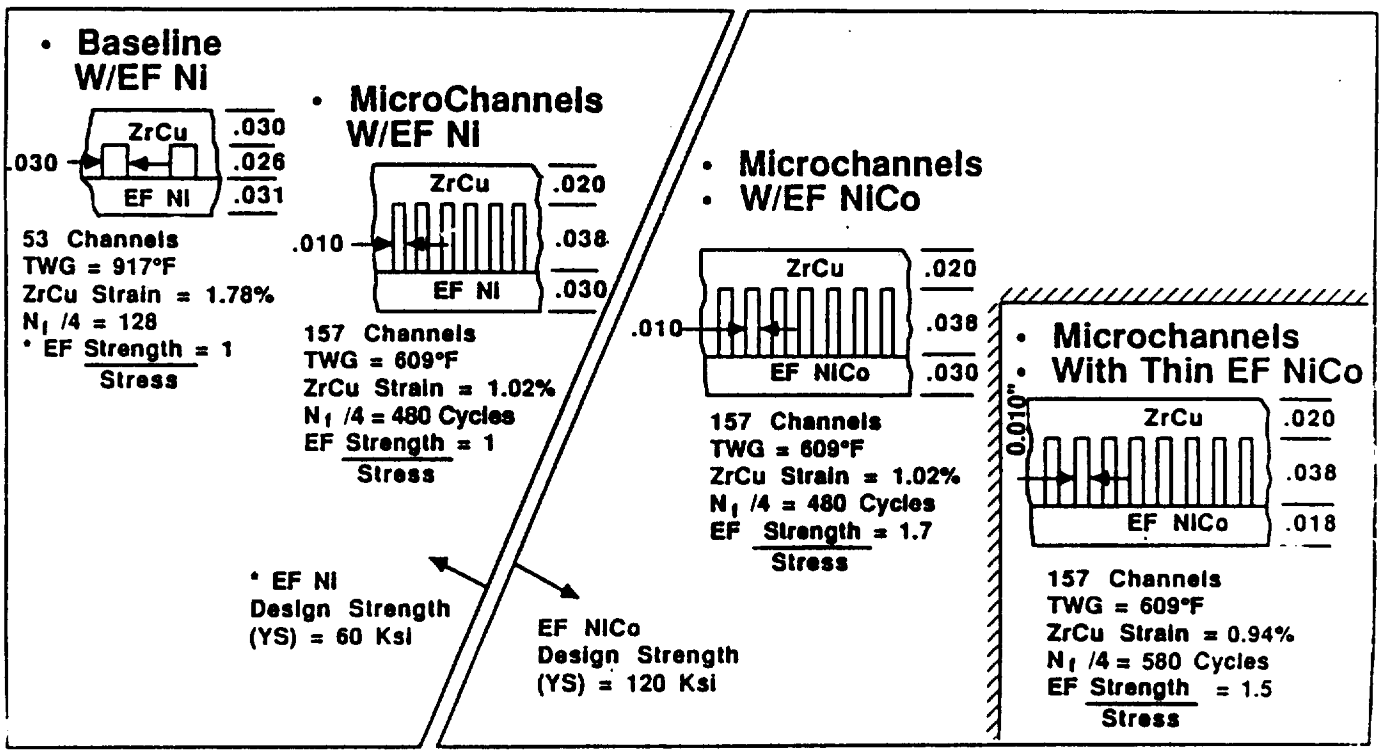


Figure 3.5.2-4 Mini Channels and Thin, Strong Closeouts Reduce Weight and Increase Life

TABLE 3.5.3-1

7.5 K LBF THRUST LEVEL TCA DESIGN CRITERIA

Injector	Same element size, spacing, & flowrate per element 180 elements (41.67 # thrust /element)
Centerbody /Regen Chamber	Scale up surface area Retain effective contraction ratio (17:1) Retain same weight flow per area (same heat flux) Retain basic channel dimensions/design

driving the oxidizer turbopump. For the 3K lbf thrust level hardware, the envelope of the cylindrical centerbody was 3.9" in diameter and was housed within the 6.8" diameter barrel of the thrust chamber (Figure 3.5.3-1). In removing the oxygen turbopump assembly from the combustion chamber, the size & weight of the regen cooled centerbody structure was reevaluated.

A remaining technical concern over the ability to adequately cool the centerbody structure with LOX was further pursued as part of this design task. The scope of the 3.0K lbf TCA program had not been expanded to demonstrate this technology prior to the increased thrust level requirements. This concern remained an issue during the 7.5K lbf TCA design task. Initial cursory heat transfer analysis undertaken in support of the 7.5K lbf TCA design task indicated potential problems with LOX as the coolant. The decision was made not to use oxygen as the coolant within the barrel of the combustion zone. Required heat pickup for the oxygen could be accomplished with a small, lightweight, heat exchanger.

Options for the TCA conceptual design were reviewed as summarized in Figure 3.5.3-2. Since a dual propellant expander cycle is baselined, balancing of the energy input into the propellants dictates the combustion chamber size. Tradeoffs between the chamber length and diameter are directly proportional to the required surface area. The main objective is to optimize the overall packaging in the smallest envelope for the lowest weight and still maintain the energy balance for the engine cycle at the design chamber pressure.

Weight concerns prompted exploration of diameter versus combustion chamber length to determine impact. The configuration labeled 'A' resembles the competition's design. Combustion of oxygen and hydrogen propellants occurs rapidly, hence a long chamber length is not required to ensure adequate mixing and combustion. An expander cycle requires a certain barrel section (high heat flux area) to provide the required heat input to the hydrogen. This is accomplished in a small diameter TCA design by increasing the L' length such that the desired hydrogen bulk temperature is reached to drive the TPA. The small diameter, long L' chamber has a definite weight advantage, however if the TCA is to fit within a restrictive envelope, the exiting area ratio is limited.

Use of a centerbody or baffle structure was recognized as a way to provide adequate heat input to the hydrogen while maximizing the achievable exit area ratio. Revising the 3.0K lbf centerbody structure, is represented in the 'C' designs. The centerbody structure denoted would not house a TPA, but would be hollow. The surface would be cooled with hydrogen.

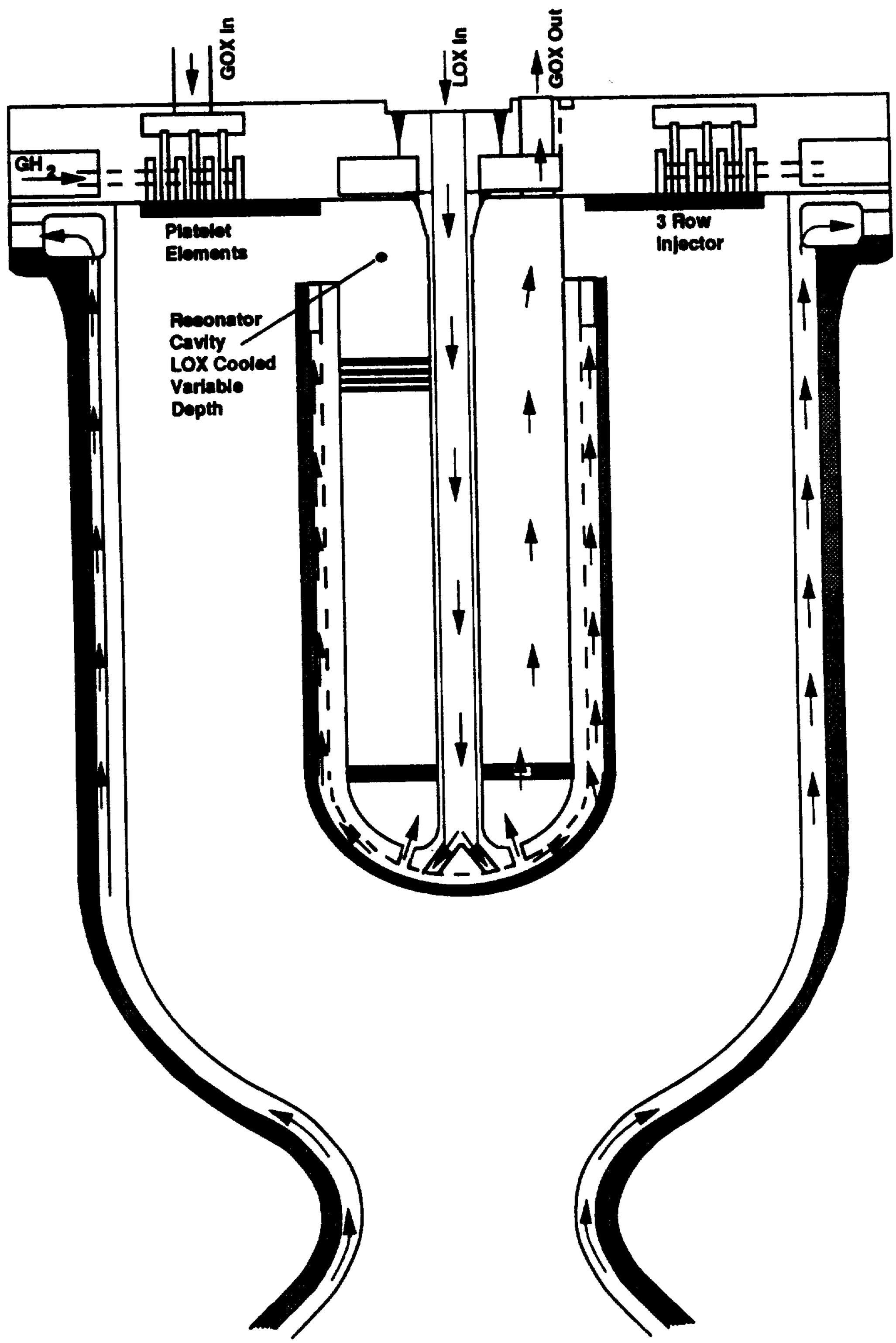
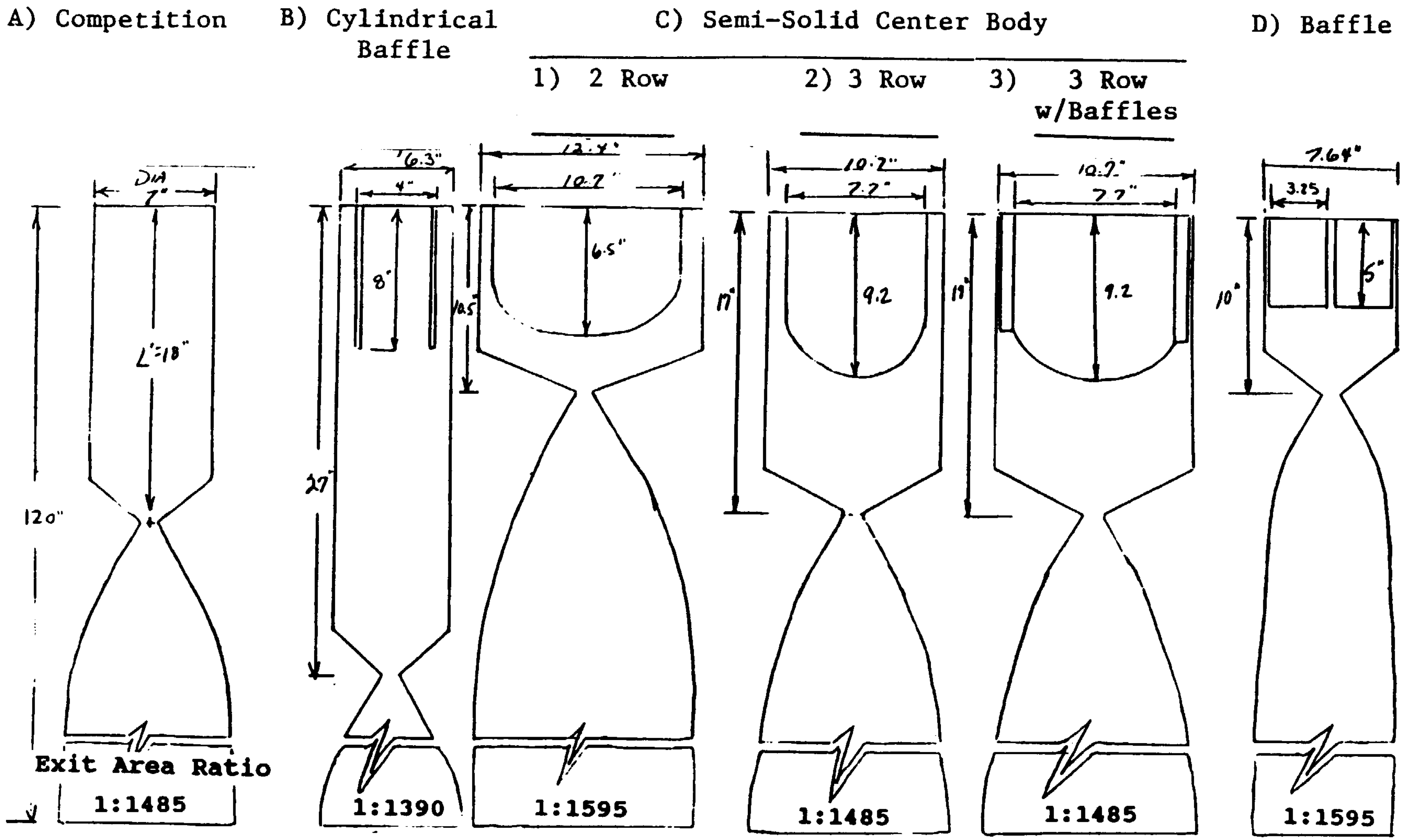


Figure 3.5.3-1 LOX Cooled Centerbody Concept



Weight:	A) Competition	B) Cylindrical Baffle	C) 1) 2 Row	C) 2) 3 Row	C) 3) 3 Row w/Baffles	D) Baffle
Outerbody	24.70	32.10	32.10	41.00	41.00	15.80
Center Body	N/A	7.90	29.90	23.80	23.80	N/A
Baffles	N/A	N/A	N/A	N/A	3.50	17.00
Total - lbs	24.70	40.00	62.00	64.80	68.30	32.80

Figure 3.5.3-2 OTV TCA Conceptual Design

Tradeoffs of injector diameter and chamber L' are summarized in the 'C1' and 'C2' design concepts. The 'C3' concept represents a midpoint between 'C1' and 'C2' in that radial baffles are added to the sides of the centerbody structure. The weight and design complexities of these concepts factored in their non-selection as a baseline design.

The semi-solid center body structure of the proposed 'C' concepts packaged a large amount of unused space. The concept labeled 'B' opens up the centerbody so that active elements can be packaged in the inner diameter. This enabled the injector diameter to be reduced, however the cylinder baffle provided very little heat transfer area and did not support shortening of the chamber L'. The 6.3 inch injector diameter of this concept represents the smallest diameter for packaging the injector elements. By splitting the injector into two cmpartments, complexity would be added to the system hydraulics. A possible benefit could be realized by using the compartmentalized injector to establish a wider throttling range.

Concept 'D', ultimately selected as the baseline design, provided the best compromise between injector diameter, chamber L', and exit area ratio with a small increase in overall TCA weight. This concept represents the use of radial, hydrogen cooled, baffles. Various baffle concepts were considered and as shown in Figure 3.5.3-3, flexibility in number and configuration ensures adequate surface area to ensure adequate hydrogen bulk temperatures.

Finalization of the TCA conceptual design enabled the power balance for the OTV engine to be completed. This established the temperature and pressure schedules to be used in the design of the coolant passages for the hydrogen cooled regen-chamber and baffles. This analysis is documented in the following sections. Section 3.5.4.3 contains the analysis methodology common to both components.

3.5.4 Analytical Design

3.5.4.1 Regenerative Cooled Chamber

3.5.4.1.1 Background

The cooling of the regeneratively cooled chamber is designed for the hydrogen coolant to enter at an exit area ratio of 28:1, flowing counter to the gas stream and exiting near the injector face.

Preliminary heat transfer analysis utilized the 3000 lbf thrust level thermal hot fire test results. A comparison of the 3000 vs the 7500 lbf thrust level designs for the throat

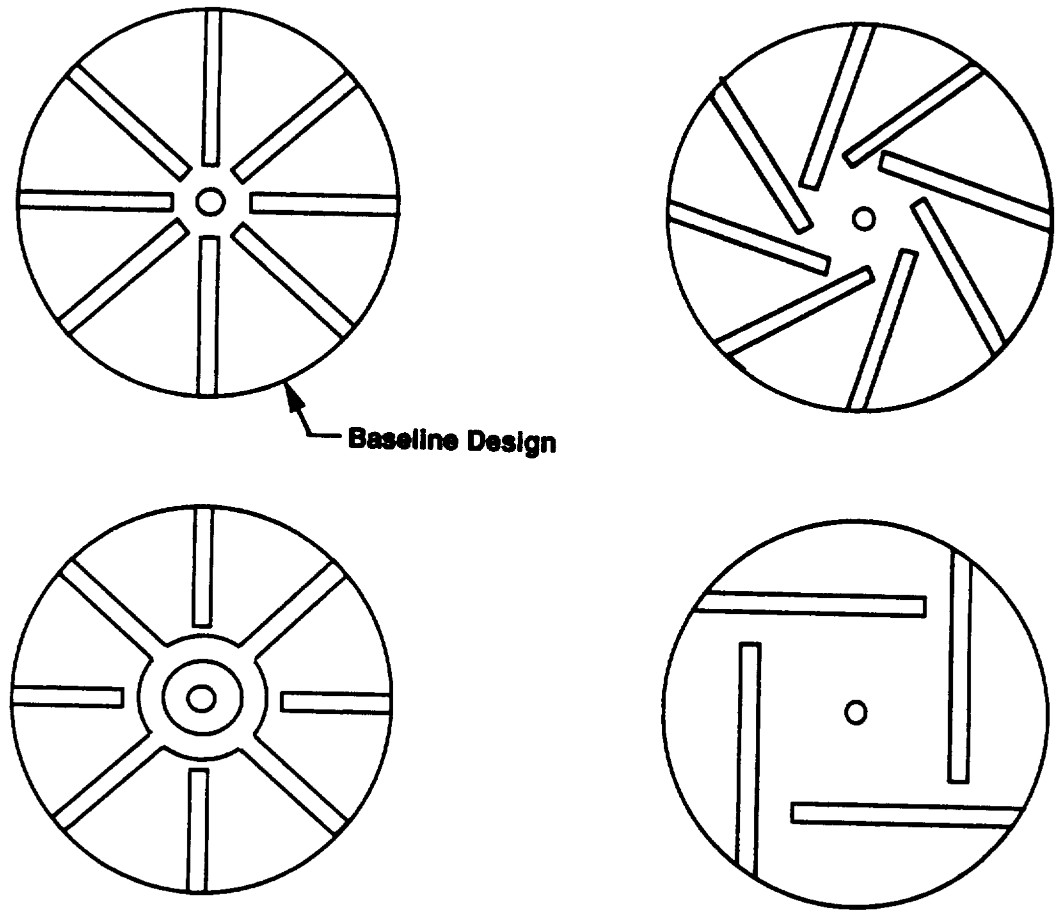


Figure 3.5.3-3 OTV Baffle Concepts

TABLE 3.5.3-2

OTV ENGINE CONCEPT SUMMARY AND COMPLEXITY COMPARISON

Item/Component	Engine Version: Cooled Centerbody	Engine Version: Baffled Injector
Oxygen Heating	Centerbody (100%)	1) O_2H_2 HEX (2/3) 2) Regen Nozzle (1/3)
Temperature Control		
• Ox Turbine Inlet	Ox Regenerator/Bypass Valve	O_2/H_2 HEX Bypass Valve
• Chamber Wall Temperature	H_2 Regenerator/Bypass Valve	H_2 Regenerator/Bypass Valve
• Oxygen Phase Change	O_2 Regenerator/Ox Back Pressure Valve	O_2/H_2 HEX, Ox BPV
Stability Devices	Acoustic Cavities	Baffles
Engine Control	Turbine Bypass Valves (2)	Turbine Bypass Valves (2)
Start Control	As Proposed Required 8 Valves	1 Idle Valve for Tank Head
• Start Valves	2 Start Valves	Start
• Back Pressure Valves	One Ox BPV	Ox and Hydrogen BPVs
• Boost Pumps	None	One on Each Circuit
Ignition	Single Flare Igniter	Dual Igniters
Regen Cooling Method		
• Throat/Chamber	87% of Hydrogen	100% of Hydrogen
• Baffles	N/A	100% of Hydrogen
• Regen Nozzle	13% of Hydrogen	100% of Oxygen

section are illustrated in Table 3.5.4-1. The initial analysis was based on a split flow (Figure 3.5.4-1) with 60% of the hydrogen flow used to cool the chamber and the remaining 40% used to cool the baffles. This hydrogen coolant flow enabled the same channel geometry used for the 3000 lbf design to be used for the 7500 lbf thrust level engine with no penalty in cycle life or pressure drop.

Manufacturing considerations prompted a closer look at how to achieve this 60/40 hydrogen split over the operating conditions. Concerns about splitting the flow were 1) possible starvation of the high heat flux throat area during transient conditions, and 2) achieving linear valve operation over all operating conditions, thereby maintaining the proper flow split. The risk was felt to be high due to channel burnout concerns involved with any imbalance of the 60% flow to cool the regen chamber. A safer and simpler approach was to use 100% of the fuel to cool the chamber. The penalty was an increase in the pressure drop with the higher mass flow rates serially through the chamber and baffle channels. Thermal analysis showed it was possible to maintain the previous pressure drop in the chamber while controlling the gas side wall temperature by controlling the depth of the channel.

Table 3.5.4-2 summaries this study on channel depth for two different bulk temperatures which demonstrate the ability to control differential temperature across the channel as a function of the coolant channel depth. As the bulk temperature is lowered, the pressure differential per inch of travel increases. For a 60/40 split at a bulk temperature of 200 degrees F, the pressure differential per inch of travel was actually higher than for 100 % flow. This is attributed to the frictional flow area which constitutes a larger percentage of the loss as the hydraulic diameter is decreased. The nominal thermal characteristics and pressure drops for the system are summarized in Table 3.5.4-3. The total system schematic (Figure 3.5.4-2) and power balances for the MR = 5, PC = 200 and 2075 psia, and MR = 7, PC = 200 and 2300 psia operating conditions are shown in Figures 3.5.4-3, -4, -5, and -6, respectively.

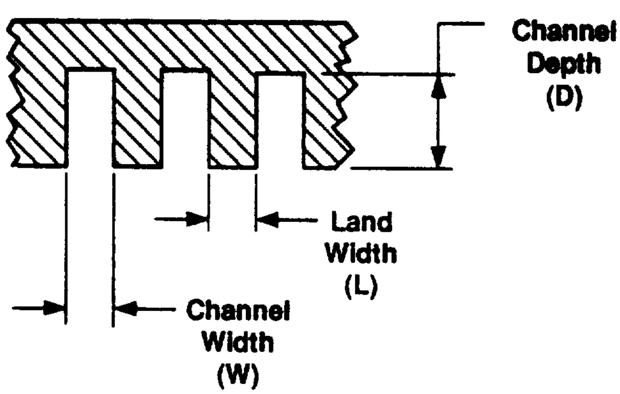
3.5.4.1.2 Parametrics

Parametric analyses were conducted at the throat plane of the OTV 7.5K lbf thrust level engine to determine cooling and pressure drop characteristics as a function of channel geometry. Since the operating condition of MR=5, PC = 2075 psia results in the highest heat flux at the throat (83 Btu/sec-in^2), the boundary conditions related to this operating point were used in the subsequent analysis. Channel depth and hydrogen bulk temperature were varied to determine the effect on gas side wall temperature, maximum temperature gradient across the coolant channel,

TABLE 3.5.4-1

COOLANT CHANNEL DESIGN COMPARISON

Thrust (lbf)	3000	7500
Gas-side wall thickness (in.)	0.02	0.02
Back-side wall thickness (in.)	0.018	0.06
Land width (in.)	0.011	0.011
Channel width (in.)	0.01	0.01
Channel depth (in.)	0.038	0.04
Gas-side material	Zr-Cu	Narloy Z
Gas-side material conductivity, ambient (Btu/in*s*F)	0.0050	0.0043
Back-side material	EFNi	EFNiCo
Back-side material conductivity, ambient (Btu/in*s*F)	0.00092	0.00014
Max. gas-side wall temperature (F)	609	699
Max. delta temp. across channel	811	816
Cycle life (Nf/4)	620	600



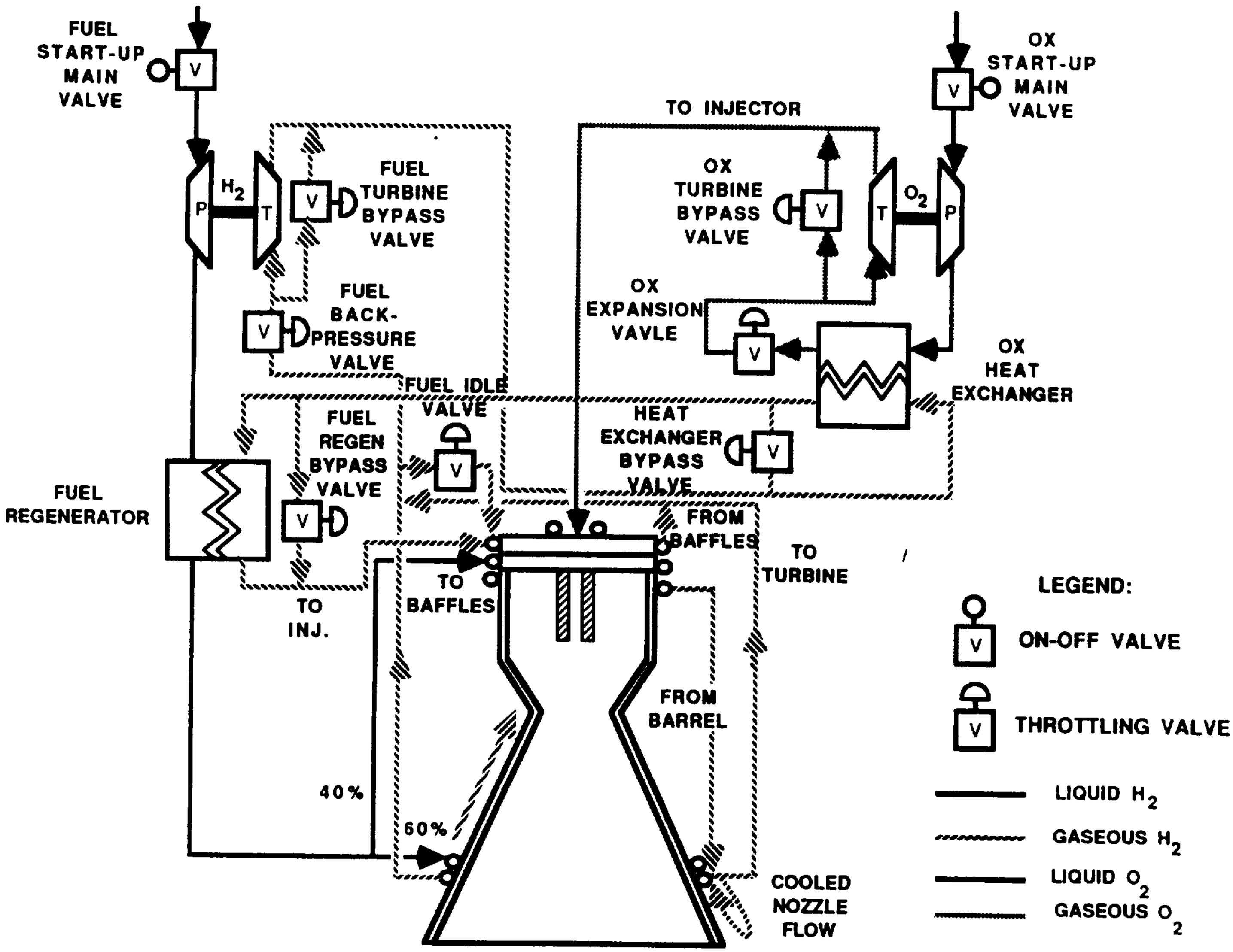


Figure 3.5.4-1 Hydrogen Cooled Regen Nozzle Schematic

TABLE 3.5.4-2

TEMPERATURE GRADIENT ACROSS CHANNEL CAN BE MAINTAINED BY VARYING CHANNEL DEPTH

	Bulk Temperature (R)/(F)	
	200/-260	400/-60
Delta T Across Channel	859	859
Channel Depth (in.)	0.065	0.092
Maximum Wall Temp (F)	610	800
Delta P per inch of travel (psia/in.)	420	320
Hydrogen Velocity (ft/sec)	760	890

Channel Width = .01", Land Width = .011", Wall Thickness = .02",
Close-out Thickness = .02", Gas side material = Narloy Z,
Close-out Material = Haynes 188, 233 Channels, 100% Hydrogen Flow,
MR = 5, Pc = 2075 psia

Table 3.5.4-3
Thrust Chamber Critical Temperatures and Pressures

	Engine Operating Condition				
	MR = 5 Pc = 200 (psia)	MR = 7* Pc = 200 (psia)	MR = 5 Pc - 2075 (psia)	MR = 7 Pc - 2300 (psia)	Design Limit
T, Wall (F) Max, Jacket	701	659	887	960	1000
T, Wall (F) Max, Throat	-82	-195	705	723	800
T, Wall (F) Max, Baffle	1025**	1027	832	941	1000
T, (F) H2, Entering Jacket	-260	-420	-235	-210	-200 (TBD)
T, (F) H2, Entering Baffles	204	100	66	135	TBD
T, (F) H2, Exiting Baffles	818	805	440	545	1200
ΔP (psia) Regen Jacket	75	22	682	575	TBD
ΔP (psia) Baffle	20	11	83	75	TBD
ΔP (psia) Total	95	33	765	650	Power Balance Dependent

*Temperatures without a Hydrogen Regenerator

**Exceeds limit at off MR, Throttled Condition

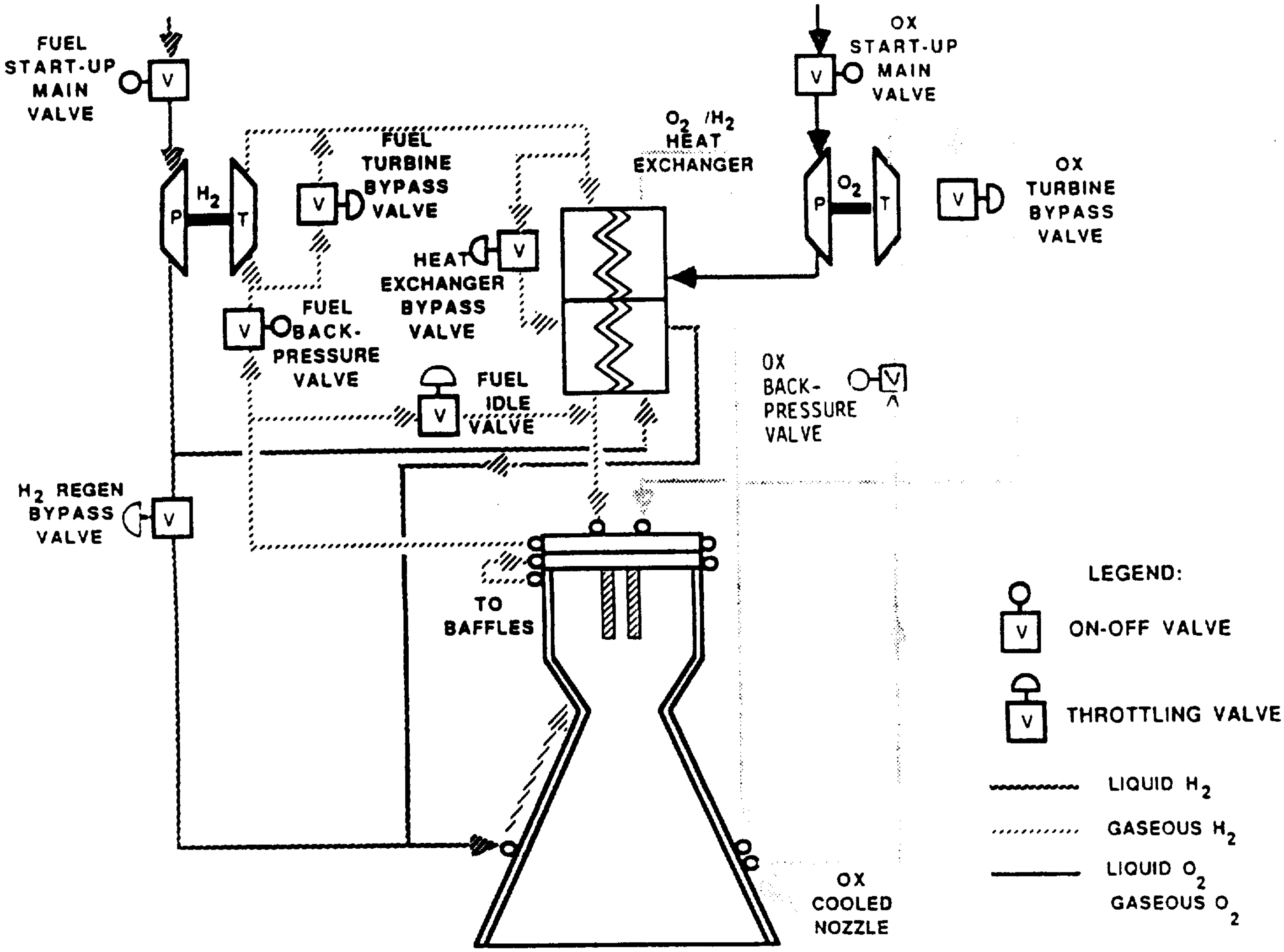


Figure 3.5.4-2 Oxygen Cooled Regen Nozzle Schematic

OTV Ox Cooled Regen Nozzle with Back Pressure Valve

Chamber Pressure (psia):	200
Mixture Ratio:	5.0
Total Flow Rate (lb/sec):	1.541
Oxidizer Flow Rate (lb/sec):	1.3
Fuel flow Rate (lb/sec)	0.3

PARAMETER	UNITS	OX	FUEL
Pump Pressure Rise	psia	301.67	382.78
Density	lb/cu ft	71.22	4.34
Pump Flow Rate	lb/sec	1.28	0.26
Pump Efficiency		0.27	0.45
Bearing Cooling Loss	% SHPpump	5.00	5.00
Boost Pump Power	% SHPpump	4.30	0.00
Head Rise	feet	609.99	12712.25
Specific Heat	BTU/lb R	0.24	3.54
Turbine Flow Rate	lb/sec	0.39	0.10
Percent Turbine Bypass		69.65	62.96
Turbine Efficiency		0.52	0.48
Pressure Ratio		1.43	1.19
Gamma		1.40	1.40
Turbine Inlet Temperature	deg R	860.0	1280.00
PUMP POWER	HP	5.77	14.01
TURBINE POWER	HP	5.77	14.01
Power Ratio (HPturb/HPpump)		1.000	1.000

Pressure Schedule

Location	OX	FUEL
Chamber Pressure	200	200
Injector Pressure Drop	30	20
Ox Heat Exchanger Exit Pressure (fuel side)	0	220
Ox HEX Pressure Drop (fuel side)	0	10
Turbine Exit Pressure	230	230
Turbine Pressure Drop	99	43
Turbine Inlet Pressure	329	273
Back Pressure Valve Pressure Drop	0	80
Cooled Nozzle Exit Pressure	329	353
Cooled Nozzle Pressure Drop	9	0
Cooled Nozzle Inlet Pressure	338	353
Baffles Pressure Drop	0	25
Baffles Inlet Pressure	338	378
Regen Jacket Pressure Drop	0	25
Ox HEX Exit Pressure (ox side)	338	403
Ox HEX Pressure Drop (ox side)	26	0
Pump Exit Pressure	364	403
Pump Pressure Increase	302	383
Pump Inlet Pressure	62	20

Figure 3.5.4-3 System Power Balance Summary at MR = 5 and Pc = 200 psia - Oxygen Cooled Regen Nozzle

OTV Ox Cooled Regen Nozzle

Chamber Pressure (psia):	2075
Mixture Ratio:	5.0
Total Flow Rate (lb/sec):	15.79
Oxidizer Flow Rate (lb/sec):	13.2
Fuel Flow Rate (lb/sec)	2.6

PARAMETER	UNITS	OX	FUEL
Pump Pressure Rise	psia	4122.36	5710.83
Density	lb/cu ft	71.22	4.34
Pump Flow Rate	lb/sec	13.16	2.63
Pump Efficiency		0.64	0.59
Bearing Cooling Loss	% SHPpump	5.00	5.00
Boost Pump Power	% SHPpump	4.30	0.00
Head Rise	feet	8335.60	189658.64
Specific Heat	BTU/lb R	0.24	3.54
Turbine Flow Rate	lb/sec	12.35	2.50
Percent Turbine Bypass		6.17	4.92
Turbine Efficiency		0.73	0.73
Pressure Ratio		1.63	2.14
Gamma		1.40	1.40
Turbine Inlet Temperature	deg R	860.0	900.00
PUMP POWER	HP	343.17	1610.88
TURBINE POWER	HP	343.17	1610.88
Power Ratio (HPturb/HPpump)		1.000	1.000

Pressure Schedule

Location	OX	FUEL
Chamber Pressure	2075	2075
Injector Pressure Drop	311	208
Ox Heat Exchanger Exit Pressure (fuel side)	2386	2283
Ox HEX Pressure Drop (fuel side)	0	45
Turbine Exit Pressure	2386	2328
Turbine Pressure Drop	1508	2660
Turbine Inlet Pressure	3894	4988
Cooled Nozzle Pressure Drop	95	0
Cooled Nozzle Inlet Pressure	3989	4988
Baffles Pressure Drop	0	371
Baffles Inlet Pressure	3989	5359
Regen Jacket Pressure Drop	0	372
Ox HEX Exit Pressure (ox side)	3894	0
Ox HEX Pressure Drop (ox side)	290	0
Pump Exit Pressure	4184	5731
Pump Pressure Increase	4122	5711
Pump Inlet Pressure	62	20

Figure 3.5.4-4 System Power Balance Summary at MR = 5 and Pc = 2075 psia - Oxygen Cooled Regen Nozzle

OTV Ox Cooled Regen Nozzle with Back Pressure Valve

Chamber Pressure (psia):	200
Mixture Ratio:	7.0
Total Flow Rate (lb/sec):	1.656
Oxidizer Flow Rate (lb/sec):	1.4
Fuel Flow Rate (lb/sec)	0.2

PARAMETER	UNITS	OX	FUEL
Pump Pressure Rise	psia	297.30	379.80
Density	lb/cu ft	71.22	4.34
Pump Flow Rate	lb/sec	1.45	0.21
Pump Efficiency		0.31	0.40
Bearing Cooling Loss	% SHPpump	5.00	5.00
Boost Pump Power	% SHPpump	4.30	0.00
Head Rise	feet	601.15	12613.28
Specific Heat	BTU/lb R	0.24	3.54
Turbine Flow Rate	lb/sec	0.39	0.10
Percent Turbine Bypass		72.89	52.75
Turbine Efficiency		0.52	0.48
Pressure Ratio		1.41	1.16
Gamma		1.40	1.40
Turbine Inlet Temperature	deg R	860.0	1280.00
PUMP POWER	HP	5.61	12.49
TURBINE POWER	HP	5.61	12.49
Power Ratio (HPturb/HPpump)		1.000	1.000

Pressure Schedule

Location	OX	FUEL
Chamber Pressure	200	200
Injector Pressure Drop	30	20
Ox Heat Exchanger Exit Pressure (fuel side)	0	220
Ox HEX Pressure Drop (fuel side)	0	10
Turbine Exit Pressure	230	230
Turbine Pressure Drop	94	37
Turbine Inlet Pressure	324	267
Back Pressure Valve Pressure Drop	0	100
Cooled Nozzle Exit Pressure	324	367
Cooled Nozzle Pressure Drop	9	0
Cooled Nozzle Inlet Pressure	333	367
Baffles Pressure Drop	0	16
Baffles Inlet Pressure	333	383
Regen Jacket Pressure Drop	0	17
Ox HEX Exit Pressure (ox side)	333	400
Ox HEX Pressure Drop (ox side)	26	0
Pump Exit Pressure	359	400
Pump Pressure Increase	297	380
Pump Inlet Pressure	62	20

Figure 3.5.4-5 System Power Balance Summary at MR = 7 and Pc = 200 psia - Oxygen Cooled Regen Nozzle

OTV Ox Cooled Regen Nozzle

Chamber Pressure (psia):	2300
Mixture Ratio:	7.0
Total Flow Rate (lb/sec):	18.558
Oxidizer Flow Rate (lb/sec):	16.2
Fuel Flow Rate (lb/sec)	2.3

PARAMETER	UNITS	OX	FUEL
Pump Pressure Rise	psia	5118.03	5325.20
Density	lb/cu ft	71.22	4.34
Pump Flow Rate	lb/sec	16.24	2.32
Pump Efficiency		0.63	0.60
Bearing Cooling Loss	% SHPpump	5.00	5.00
Boost Pump Power	% SHPpump	4.30	0.00
Head Rise	feet	10348.88	176851.66
Specific Heat	BTU/lb R	0.24	3.54
Turbine Flow Rate	lb/sec	16.08	2.30
Percent Turbine Bypass		4.01	3.48
Turbine Efficiency		0.73	0.73
Pressure Ratio		1.85	1.84
Gamma		1.40	1.40
Turbine Inlet Temperature	deg R	860.0	1000.00
PUMP POWER	HP	533.31	1308.62
TURBINE POWER	HP	550.05	1342.25
Power Ratio (HPturb/HPpump)		1.031	1.026

Pressure Schedule

Location	OX	FUEL
Chamber Pressure	2300	2300
Injector Pressure Drop	345	230
Ox Heat Exchanger Exit Pressure (fuel side)	2645	2530
Ox HEX Pressure Drop (fuel side)	0	50
Turbine Exit Pressure	2645	2580
Turbine Pressure Drop	2235	2167
Turbine Inlet Pressure	4880	4747
Cooled Nozzle Pressure Drop	100	0
Cooled Nozzle Inlet Pressure	4980	4747
Baffles Pressure Drop	0	299
Baffles Inlet Pressure	4980	5046
Regen Jacket Pressure Drop	0	299
Ox HEX Exit Pressure (ox side)	4880	0
Ox HEX Pressure Drop (ox side)	300	0
Pump Exit Pressure	5180	5345
Pump Pressure Increase	5118	5325
Pump Inlet Pressure	62	20

Figure 3.5.4-6 System Power Balance Summary at MR = 7 and Pc = 2300 psia - Oxygen Cooled Regen Nozzle

hydrogen velocity, and pressure drop per inch of coolant travel. A bulk temperature range from 150 to 400 degree R and a channel depth range from 0.02 to 0.20 inches was studied for a channel width = 0.01", land width = 0.011", wall thickness = 0.02", and close-out thickness = 0.02". Thermal conductivity properties used include those for NASA- Z (a Cu/Ag/Zr copper alloy) for the gas side wall material, and Haynes 188 for the close-out material (to approximate NiCo). The total hydrogen mass flow of the system was used in the evaluation.

The analysis indicated that the maximum gas side wall temperature becomes less sensitive to the bulk temperature as the channel depth is increased. The increase in channel depth also results in a decrease in coolant velocity which reduces the pressure loss. These trends are illustrated in Figures 3.5.4-7 and 3.5.4-8, respectively. Maximum temperature gradient across the channel was shown to be more sensitive to the bulk temperature as the channel depth is increased, as shown in Figure 3.5.4-9. A comparison of these trends is illustrated in Figure 3.5.4-10.

Figures 3.5.4-11 and 3.5.4-12 illustrate the effect channel geometry has on the maximum wall temperature, maximum temperature gradient across the channel, hydrogen velocity, and pressure drop per inch of coolant travel for coolant bulk temperatures of 200 and 400 degree R, respectively. Using the low cycle fatigue (LCF) data for NASA-Z copper, a life of 500 cycles corresponds to a maximum strain of 1.7%. Using the relationship of % strain = 2 α ΔT, and a thermal coefficient of expansion of 9.89E-06 (in/in F), the required temperature gradient across the channel is 859 degrees.

Using these trends the parametric analyses were completed to define the geometry of the throat coolant channel. The operating condition evaluated was the worst case cooling condition imposed by the operating point of MR = 7 & Pc = 2300 psia. The channel geometry was optimized at the throat based on a maximum aspect ratio (channel depth to width) of 10:1, a maximum temperature gradient across the channel of 860 degrees R (cycle life of 500), and a maximum gas side wall temperature of 700 degree F. Hydrogen inlet temperatures were evaluated over the range of 100 to 500 degree R. The hydrogen entered the regen jacket at an area ratio of 28:1, flowing counter to the gas stream. A gradual transition zone (no step machining) was assumed for the coolant channel geometry. Optimization of the channel design was established to minimize thermal strain levels in the throat area thereby promoting extended cycle life. However, by decreasing the channel temperature differential a reduction in the temperature differential across the wall was also realized.

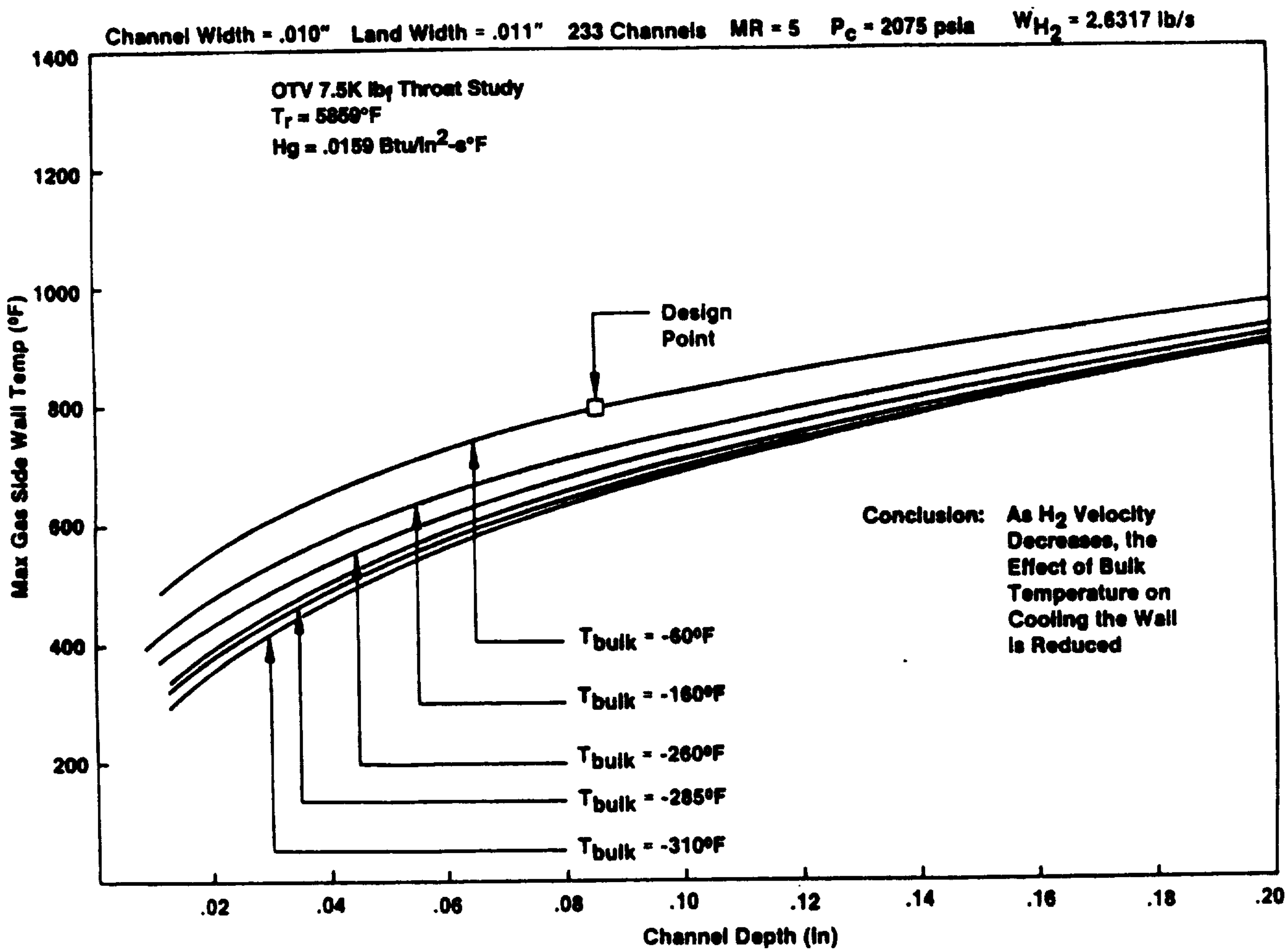


Figure 3.5.4-7 Maximum Gas Side Wall Temperature vs. Coolant Channel Depth

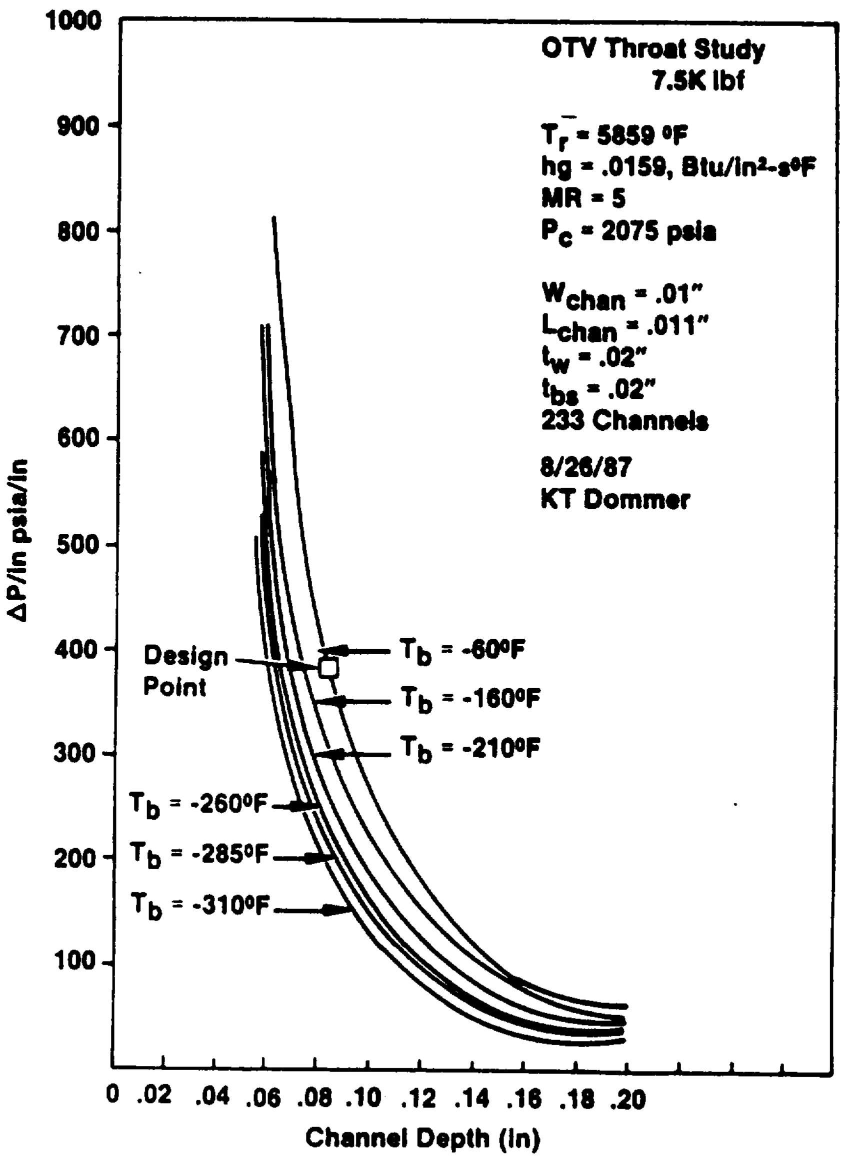


Figure 3.5.4-8 Pressure Gradient Across Channel per Inch vs. Channel Depth as a Function of Hydrogen Bulk Temperature

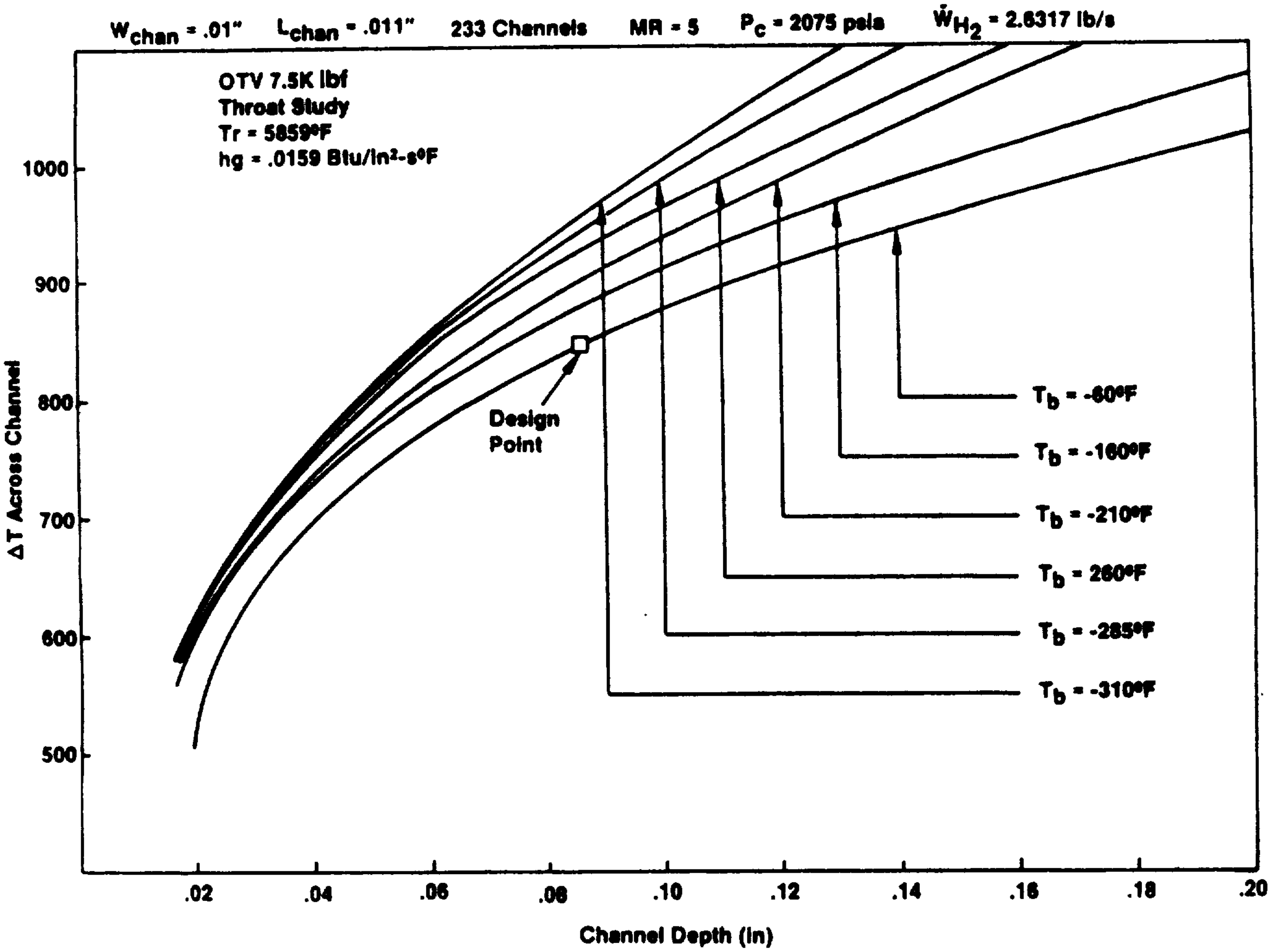


Figure 3.5.4-9 Temperature Gradient Across Coolant Channel vs. Coolant Channel Depth

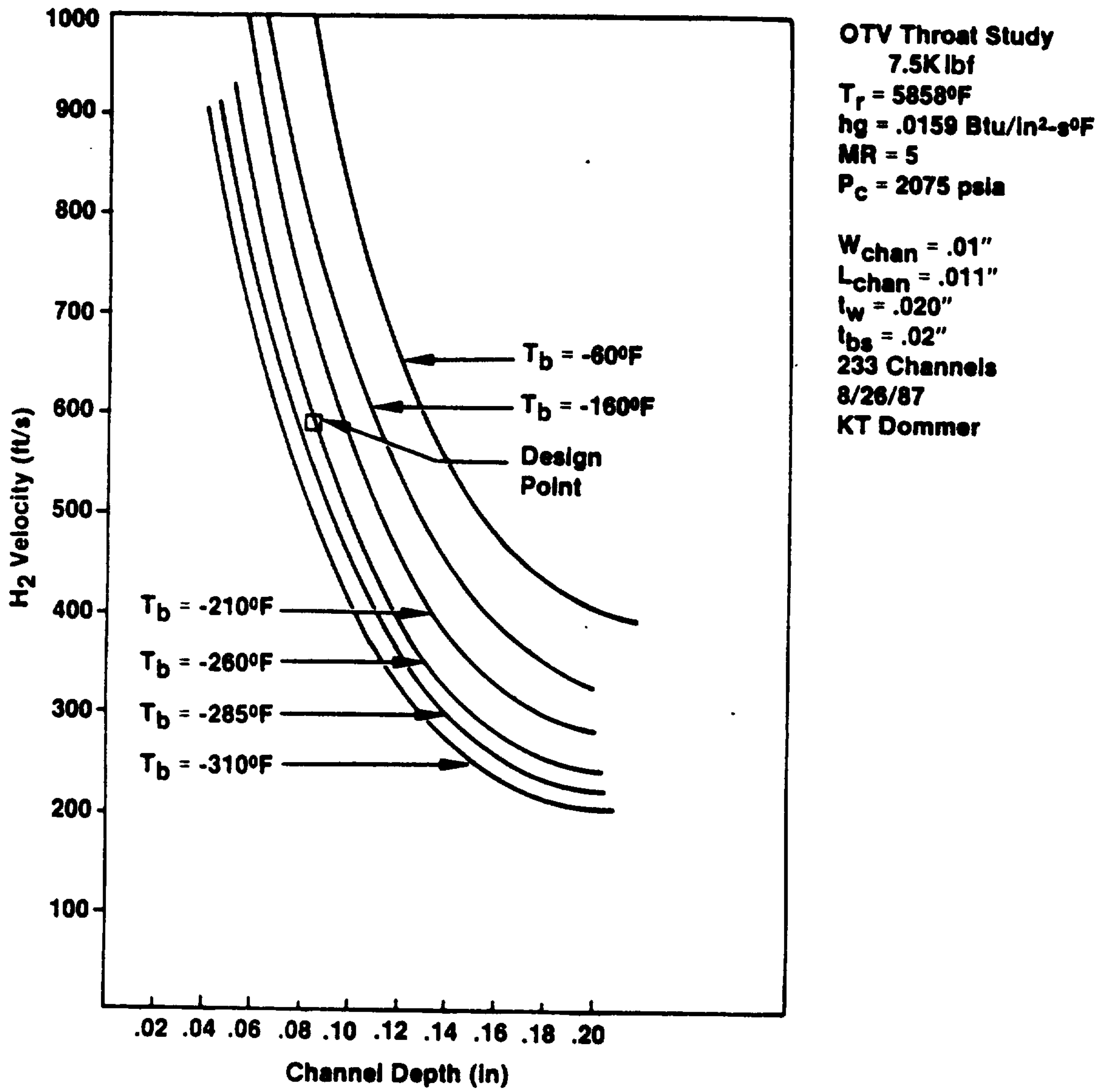


Figure 3.5.4-10 Coolant Channel Velocity vs. Coolant Channel Depth as a Function of Hydrogen Bulk Temperature

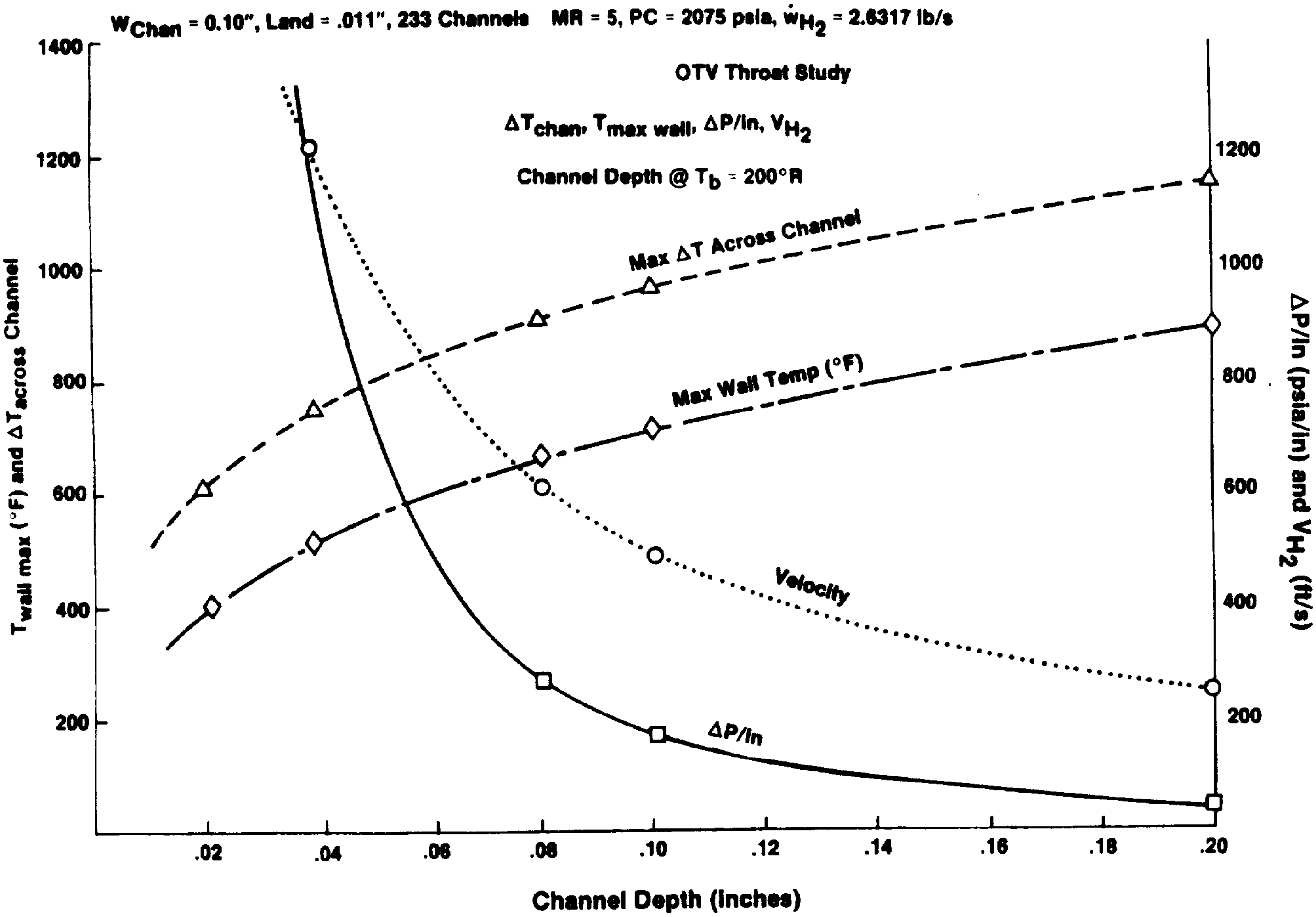


Figure 3.5.4-11 OTV Throat Study - for Hydrogen Bulk Temperature of 200R

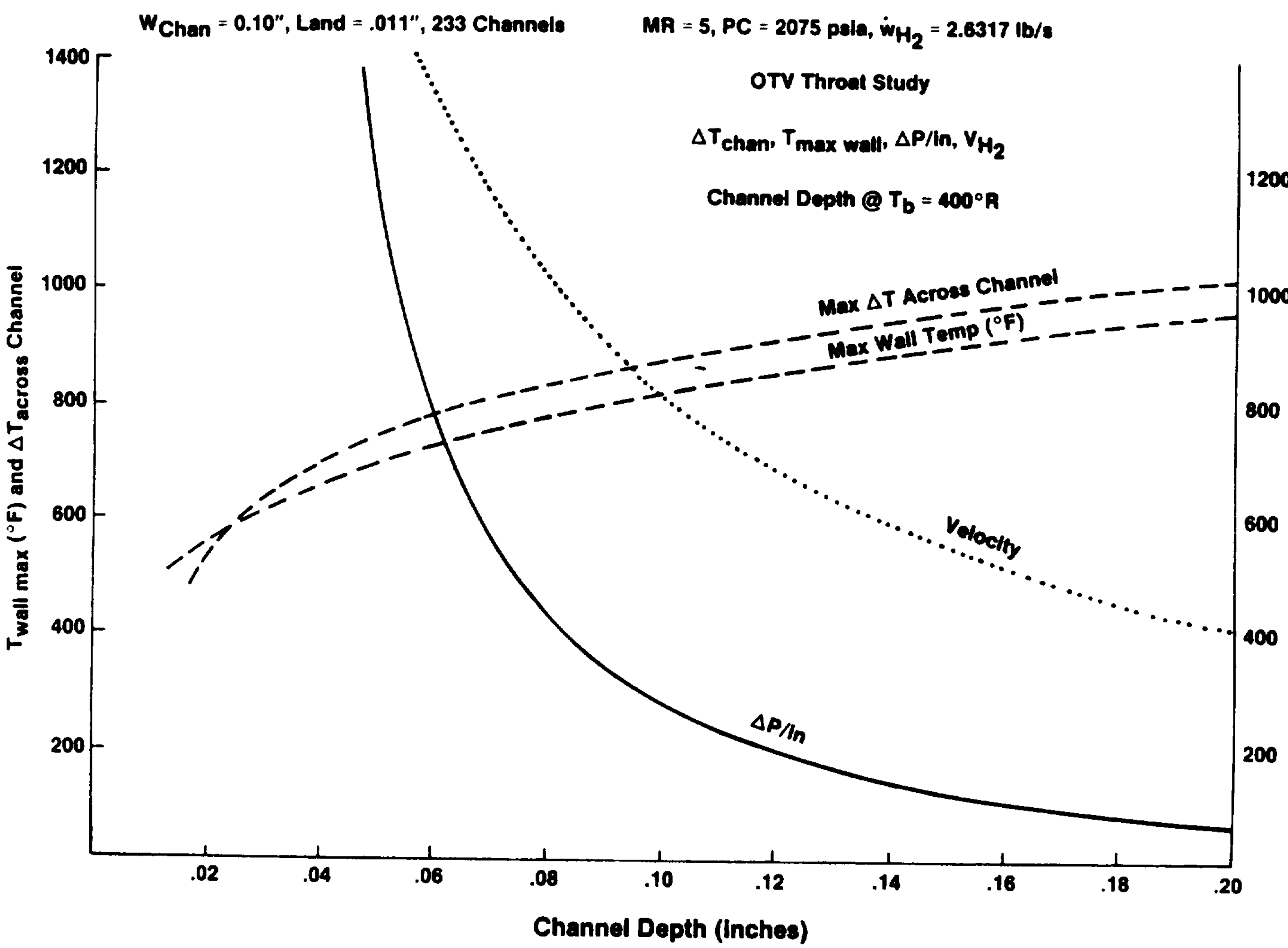


Figure 3.5.4-12 OTV Throat Study - for Hydrogen Bulk Temperature of 400 R

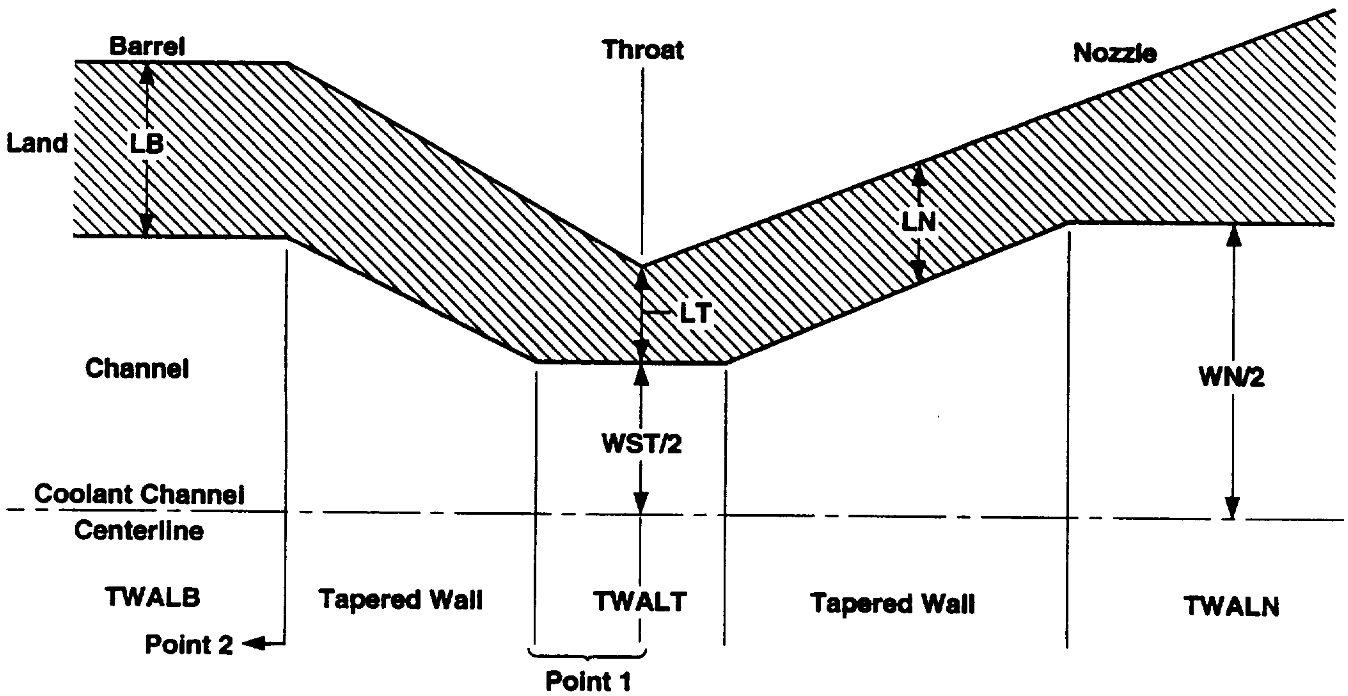
Figure 3.5.4-13 illustrates the channel and land width definitions. Figures 3.5.4-14 and 3.5.4-15 show expected coolant pressure drops from an exit area ratio of 28:1 to the throat for various nozzle channel depths and land widths at hydrogen inlet temperatures of 120 & 300 degree R, respectively. The pressure drop to the throat becomes fairly insensitive to nozzle channel widths above 0.025". When the nozzle land width is increased the high velocity associated with the throat channel geometry is maintained for a longer distance resulting in a pressure drop increase.

Figure 3.5.4-16 illustrates the pressure drop expected through the entire regen cooled jacket as a function of hydrogen inlet temperature for several barrel coolant channel geometries. A cross-plot of the same information but as a pressure drop versus barrel coolant channel geometries for three hydrogen inlet temperatures is shown in Figure 3.5.4-17. The nozzle land width assumed was the optimized nozzle land width of 0.011" indicated in Figures 3.5.4-14 and 3.5.4-15. The nozzle coolant channel width used in the study was 0.03". Other parameters held constant are indicated on the graphs. Figure 3.5.4-18 indicates the delta increase in pressure drop associated with various non-optimized land widths.

Figure 3.5.4-19 shows the channel depth at the throat plane for a channel width of 0.01" and hydrogen temperature exiting the regen jacket versus inlet hydrogen temperature. Also included on the chart is the effect of hydrogen inlet temperature on the maximum throat temperature associated with a delta temperature across the channel of 860 degrees R. Pressure drops for the high thrust MR = 5 operating point can be approximated by multiplying the pressure drop associated with the MR = 7 condition by 1.29, this is the square of the ratio of the hydrogen flow rate at MR = 5 to that at MR = 7.

3.5.4.1.3 Analysis Conclusions

The nominal regen-cooled jacket gas side wall temperature profile for the worst case condition (MR = 7, Pc = 2300 psia) is shown in Figure 3.5.4-20. This was the worst case in terms of wall temperature control due to the high chamber pressure and low hydrogen coolant flow rate (high mixture ratio). The maximum throat gas side wall temperature at the nominal flow rate and coolant channel geometry for this high MR, high chamber pressure condition is 723 degree F. Maximum nominal throat gas side wall temperature for the MR = 5, Pc = 2075 psia condition is 705 degree F. The low chamber pressure (200) psia maximum nominal throat gas side wall temperatures for mixture ratios of 7 & 5 are -195 and -82 degree F, respectively.



LAND WIDTH BETWEEN CHANNELS

LB – Land Width, Barrel
LT – Land Width, Throat
LN – Land Width, Nozzle

CHANNEL WIDTH

WST/2 – Channel Width, Throat
WN/2 – Channel Width, Nozzle

CLOSEOUT WALL THICKNESS

TWALB – Closeout, Barrel
TWALT – Closeout, Throat
TWALN – Closeout, Nozzle

MODELED GAGES

Point 1 – Throat Location Modeled
Point 2 – Barrel Location Modeled

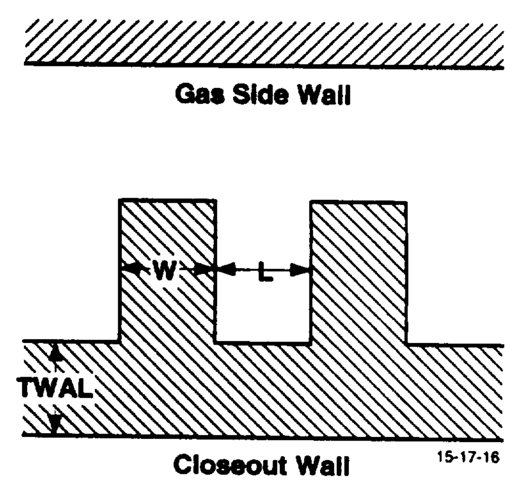


Figure 3.5.4-13 Coolant Channel Layout for Analytical Model Input

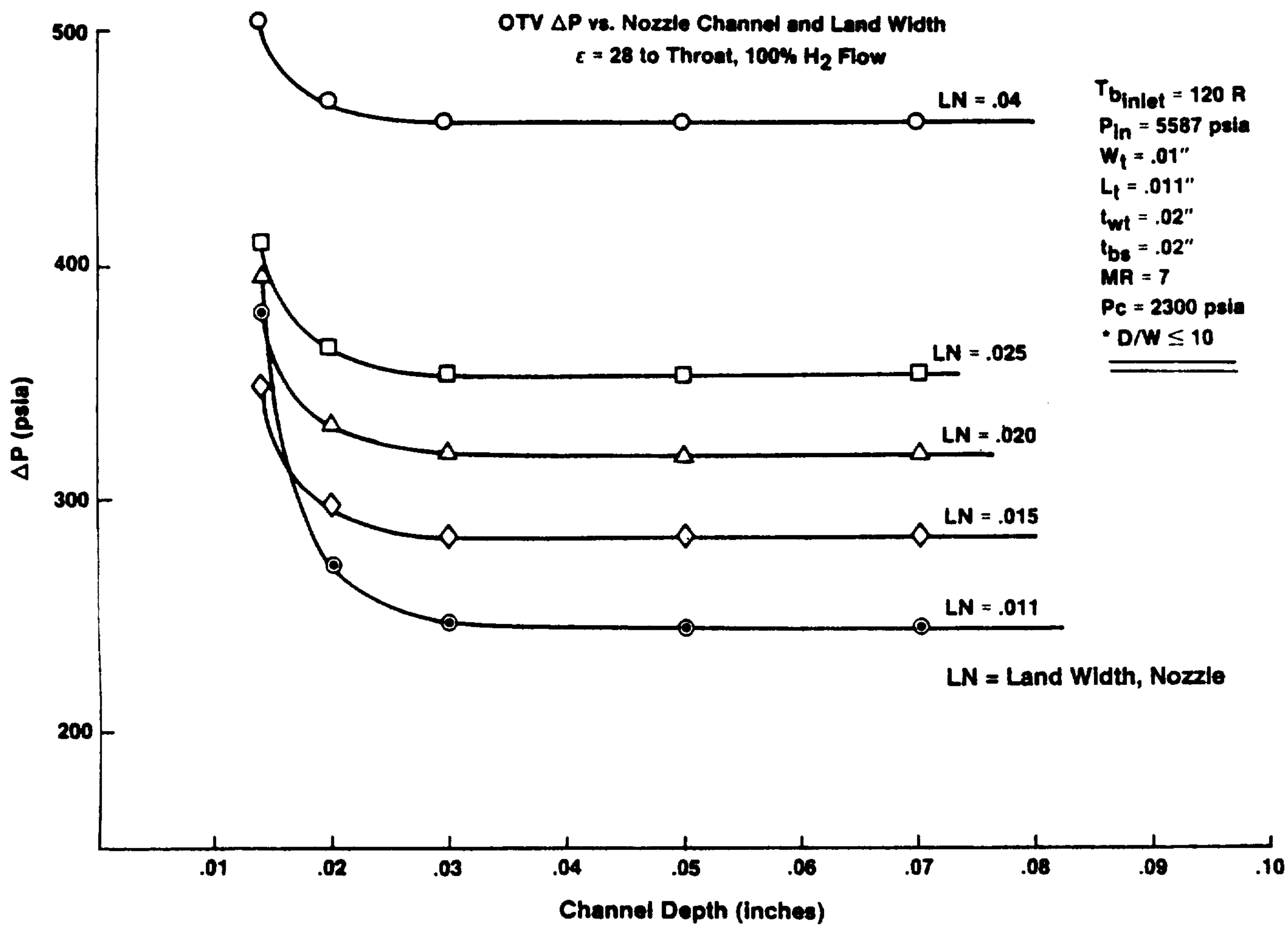


Figure 3.5.4-14 Pressure Gradient vs. Nozzle Channel Depth and Land Width

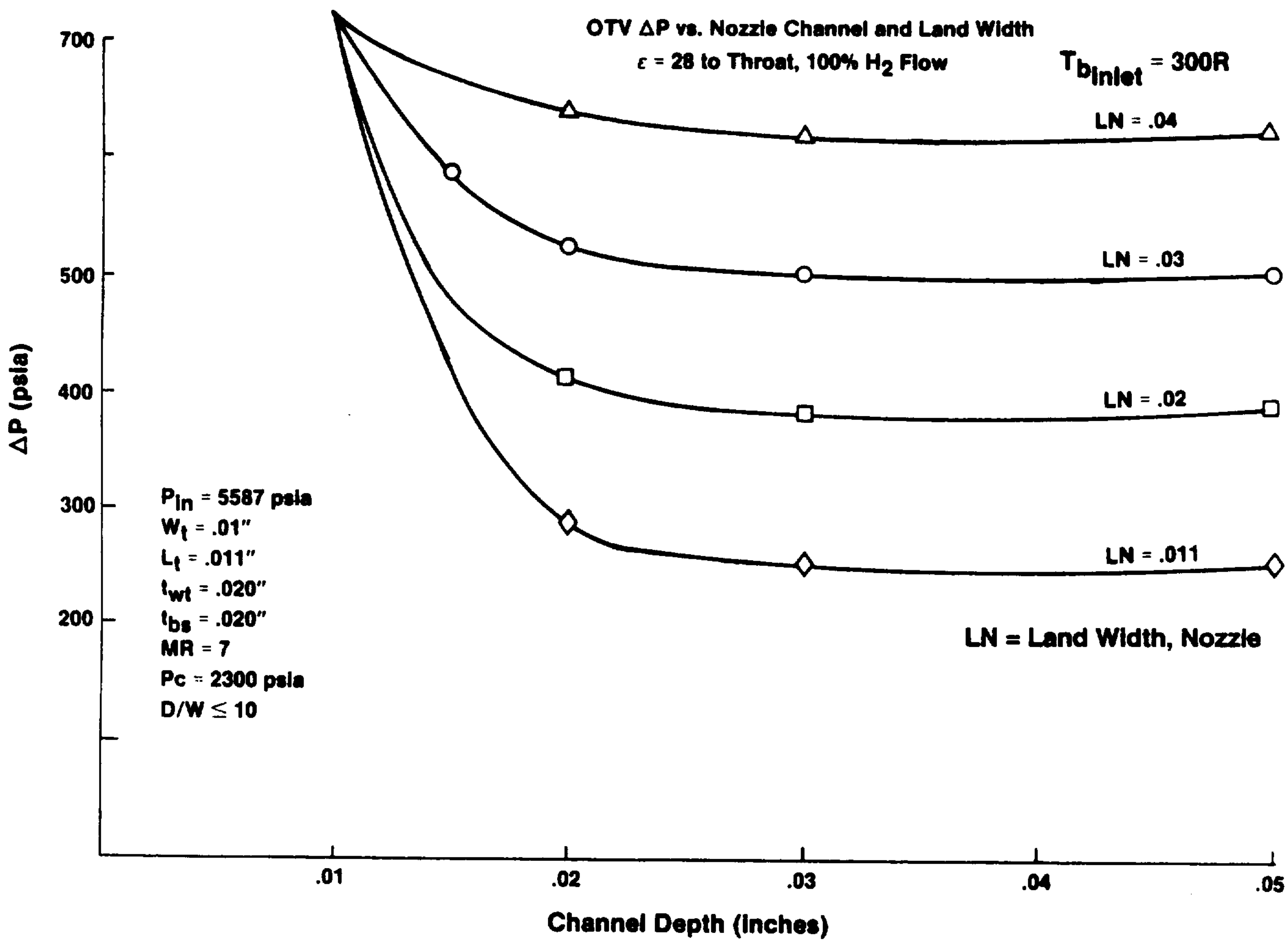


Figure 3.5.4-15 Pressure Gradient vs. Nozzle Channel Depth and Land Width (Expanded Scale)

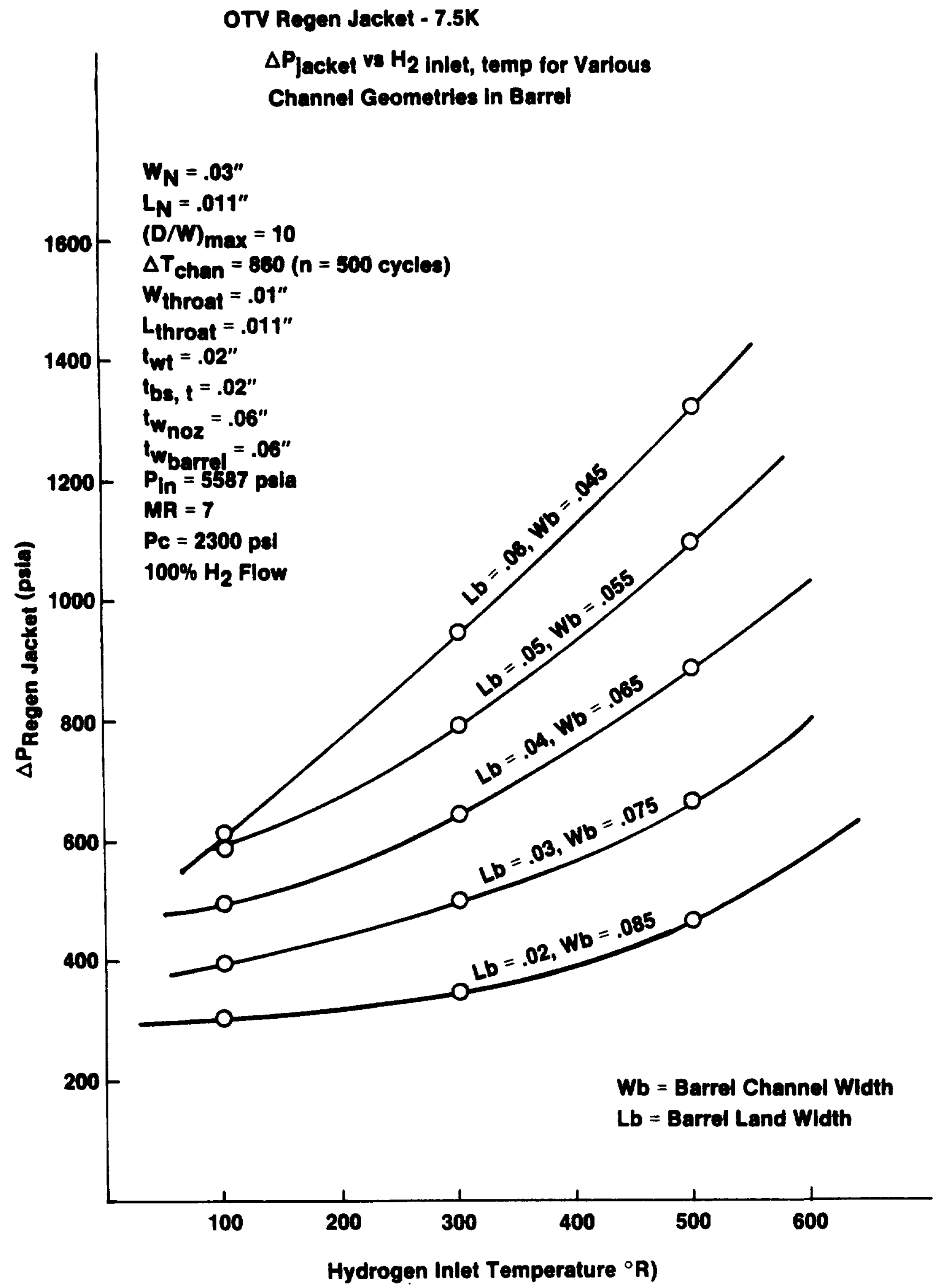


Figure 3.5.4-16 Pressure Gradient for Regen Jacket vs. Hydrogen Inlet Temperature

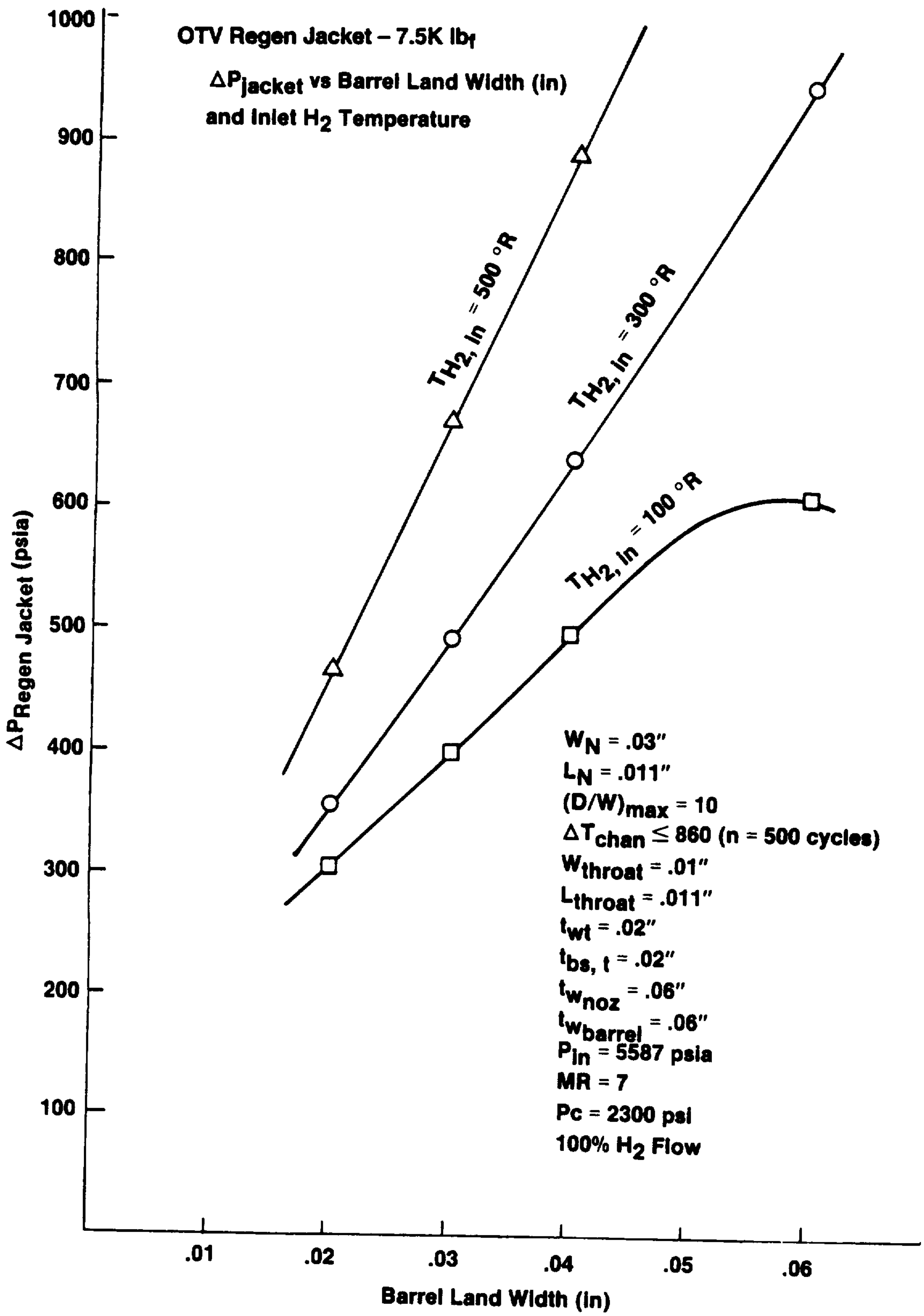


Figure 3.5.4-17 Pressure Gradient vs. Barrel Land Width

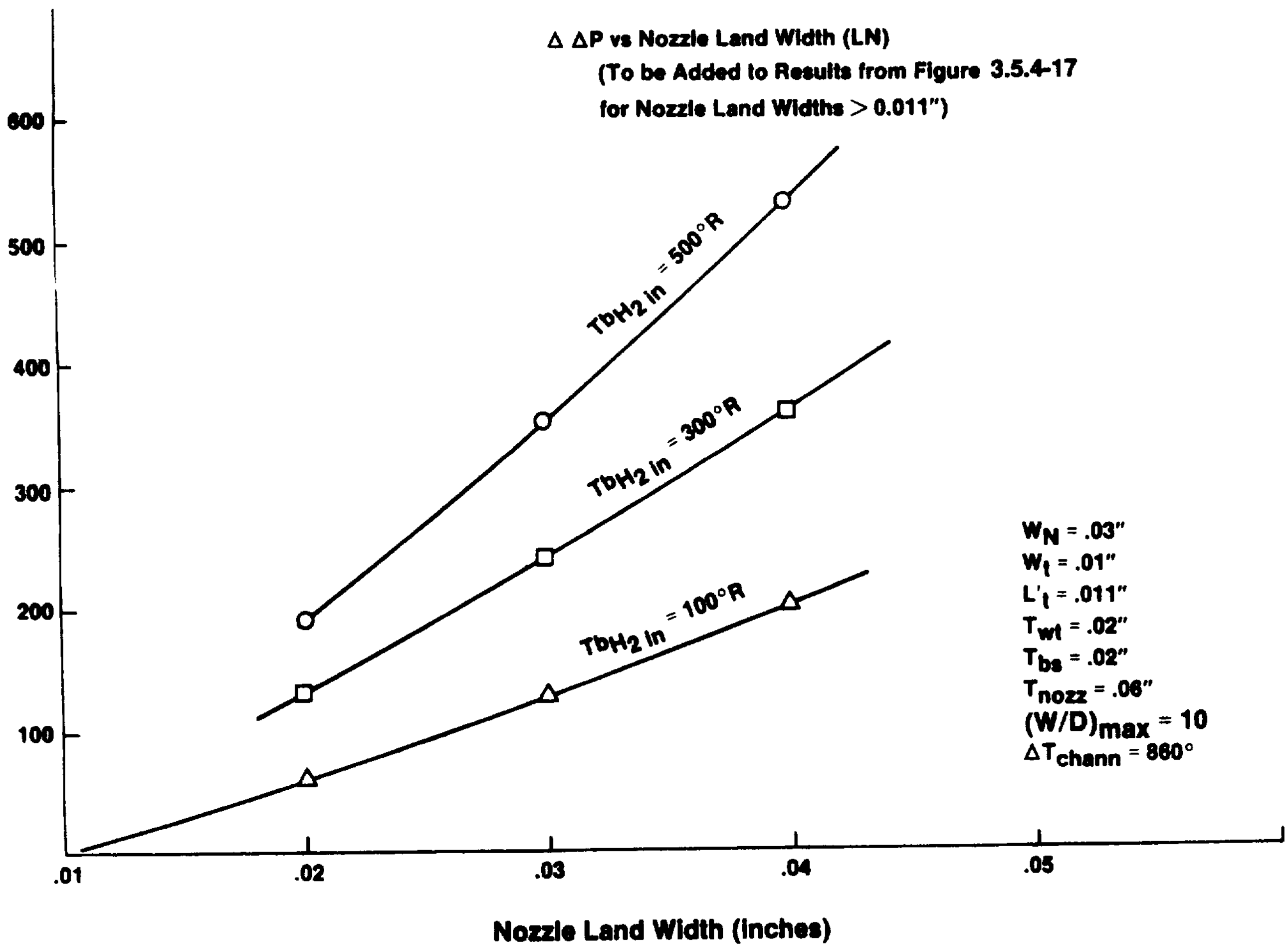


Figure 3.5.4-18 Nozzle Land Width vs. Pressure Drop for Varying Hydrogen Inlet Temperature

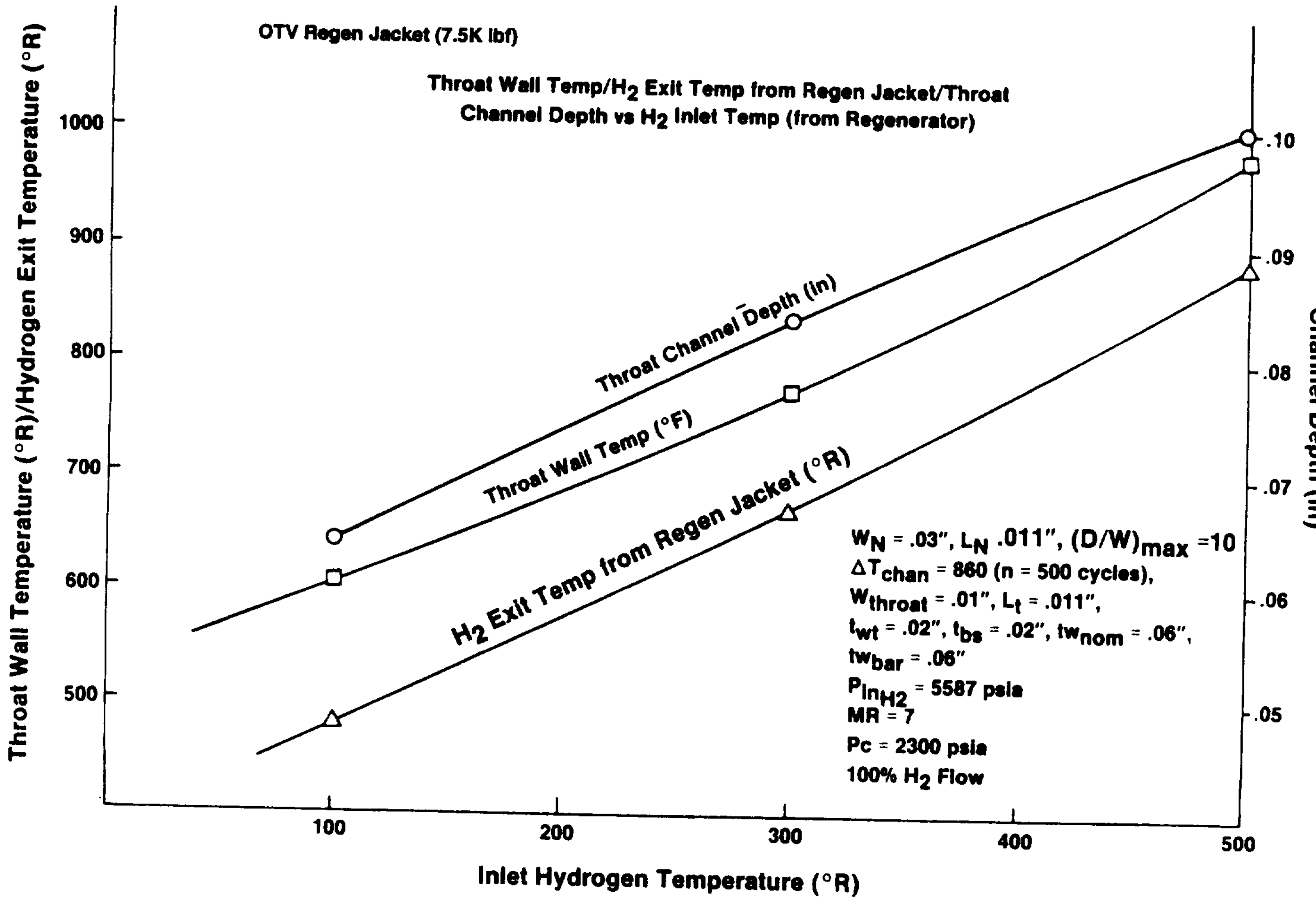


Figure 3.5.4-19 Channel Geometry Influences on Wall and Bulk Temperatures

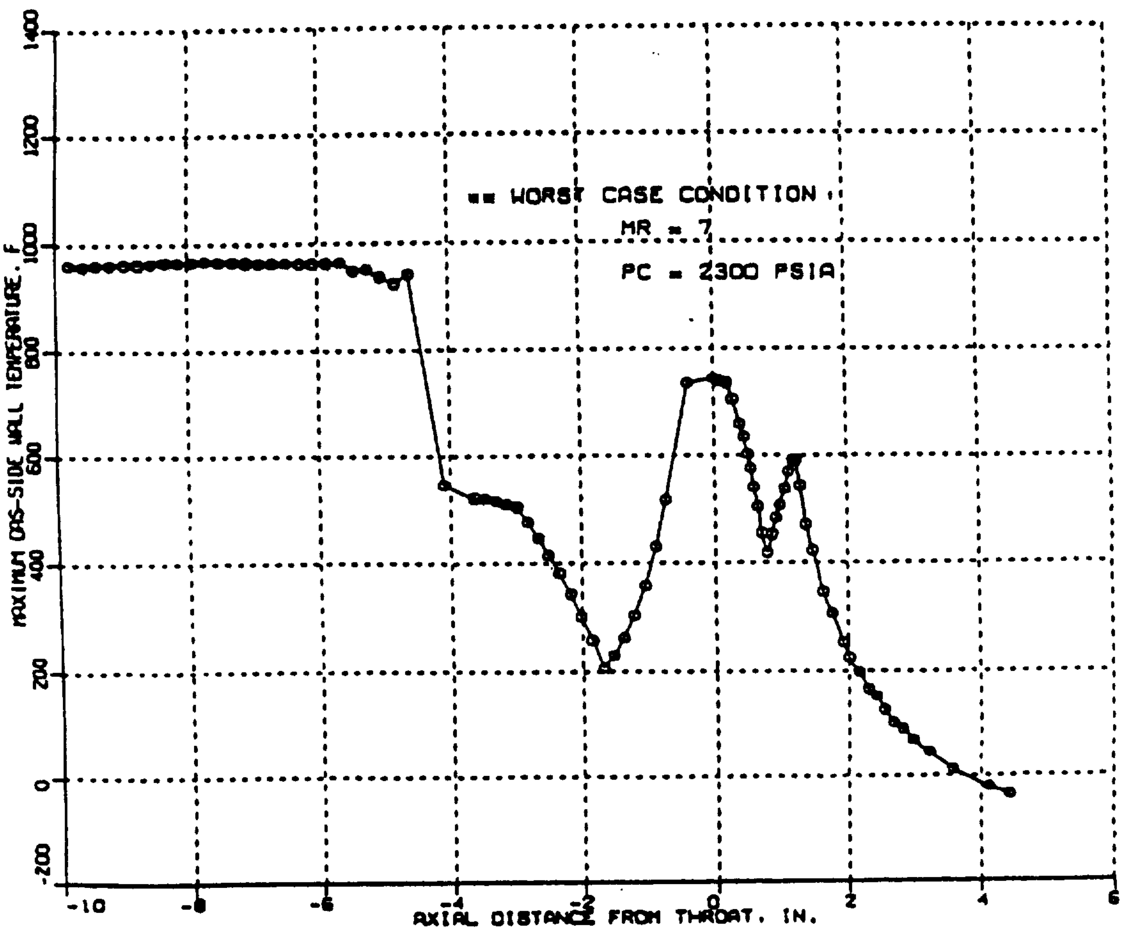


Figure 3.5.4-20 OTV 7.5K lbf Regen Jacket Nominal Gas Side Wall Temperature vs. Axial Distance From Throat (Worst Case)

Accounting for the manufacturing tolerances, the maximum gas side wall temperature at the throat region for the MR = 7, PC = 2300 psia is predicted to be 879 degree F. Figure 3.5.4-21 shows the throat wall temperature as a function of the positive tolerance range.

3.5.4.1.4 Analysis Methodology

The final design for the regen cooled jacket was based on various interacting structural, thermal, hydraulic, and fabrication considerations. The minimum channel width was set at 0.01 inches coupled with a maximum aspect ratio (channel depth to width) of 10. The minimum land width was equal to 0.011". The ability to achieve these dimensions was demonstrated on the OTV 3K lbf TCA IR&D program. To minimize the pressure drop throughout the system, an aspect ratio of 10 for the cooling channels was set as the design limit. The tolerance used in the chamber design analysis was +/- 0.002 inches.

The wall strength criteria used in designing channels is based on fully-elastic hot walls. The maximum allowable channel width to gas side wall thickness with tolerances is given as:

$$\frac{W_{max}}{tw_{min}} = 1.414\ (F_{ty}/\Delta P_{max})\ {}^{**}.5$$

where:

W_{max}	=	max channel width (in.)
tw_{min}	=	min gas side wall thickness (in.)
F_{ty}	=	ultimate yield strength (psi)
ΔP_{max}	=	max coolant channel pressure drop (psi)

The regen cooled jacket design procedure began with the assumption of the inlet H2 stagnation pressure and temperature for the MR = 7, PC = 2300 psia operating condition. For a given channel width, the wall strength criterion defines a minimum wall thickness. The selected land width and channel width at the throat plane defines the number of cooling channels required for the regeneratively cooled chamber design. The channel depth is determined based on the maximum allowable wall temperature, maximum aspect ratio, and maximum temperature gradient from the gas side to the back side of the channel to satisfy the required cycle life. The maximum delta temperature to obtain a cycle life of 500 was calculated with:

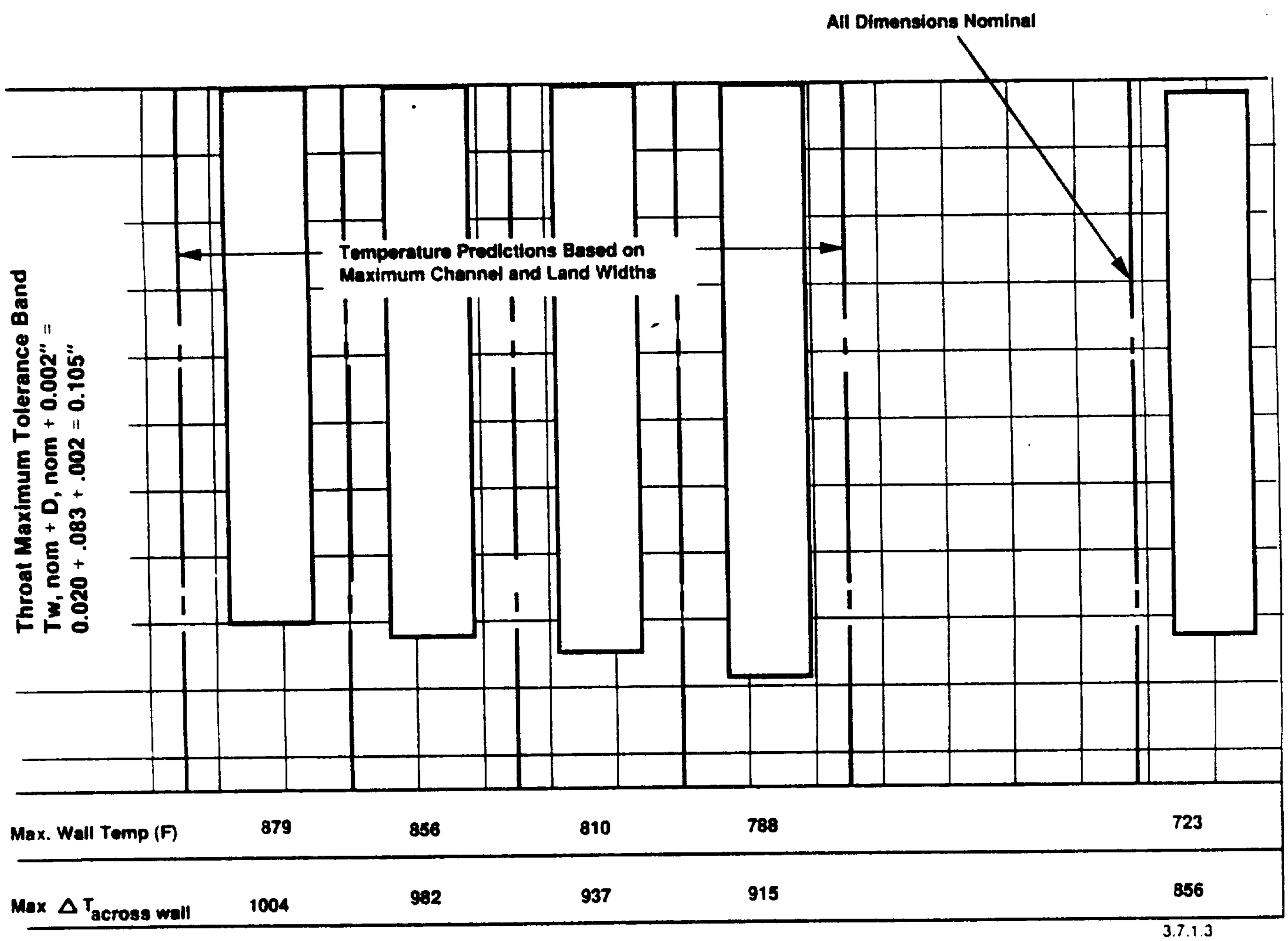


Figure 3.5.4-21 Geometric Tolerance Study of the Hydrogen Coolant Channel at the Throat

$$\Delta T_{max} = \%\ e/(2 * \alpha)$$

where:

ΔT_{max}	=	max temperature differential
$\%e$	=	percent strain
α	=	thermal coefficient of expansion

The regen cooled jacket is designed to operate at a chamber pressure range of 200 to 2300 psia for a corresponding MR range of 7 to 5. Table 3.5.4-3 provides the resultant nominal thermal and hydraulic characteristics for the four operating conditions for the proposed hardware design. Appendix A-4 contains selected resultant SINDA single channel temperature distributions. The local channel widths, lands, wall thickness, and depths are summarized in Table 3.5.4-4 and are illustrated as a function of axial distance from the throat in Figure 3.5.4-22.

3.5.4.1.4 Gas Side Boundary Conditions

The gas side heat flux profile for the high and low chamber pressure operating conditions are shown in Figures 3.5.4-23 and 3.5.4-24, respectively. The gas side heat transfer coefficient is evaluated with the Bartz correlation:

$$Nu_f = .026\ Cg\ Re_f^{.8}\ Pr_f^{.4}$$

where:

Nu_f	=	Nusselt number
Cg	=	heat transfer correlation coefficient (Bartz correlation)
Re_f	=	Reynolds Number
Pr_f	=	Prandtel Number

All properties are evaluated at a film temperature equal to the average of the wall temperature and the adiabatic wall temperature. All property and temperature data assumed a fully mixed gas and were obtained from the TRAN 72 computer program. The Cg profile was developed from the OTV 3K lbf thrust level program. A discussion follows.

The Cg profile used to calculate the gas side heat transfer coefficients in the barrel section are based on the test results of OTV 3K lbf test #2459-120-A6-147.

Table 3.5.4-4

Summary – Regen Jacket Channel Geometry

POSITION ON CHAMBER	AREA RATIO A/AT	COOLANT PASSAGE DEPTH (IN.) D	COOLANT PASSAGE WIDTH (IN.) W	LAND WIDTH (IN.) L	WALL THICKNESS (IN.)
AFT INJECTOR FACE	25.32	0.105	0.065	0.040	0.0600
END OF BAFFLES	25.32	0.400	0.062	0.040	0.0582
MID CONVERGING SECTION	12.20	0.287	0.029	0.040	0.0337
THROAT	1.00	0.083	0.010	0.011	0.0200
AREA RATIO 2	2.01	0.190	0.019	0.011	0.0380
AREA RATIO 4	4.03	0.300	0.030	0.013	0.0600
AREA RATIO 8	8.06	0.300	0.030	0.030	0.0600
AREA RATIO 17	16.90	0.300	0.030	0.056	0.0600
HYDROGEN INLET MANIFOLD	28.00	0.300	0.030	0.080	0.0600

15-17-15

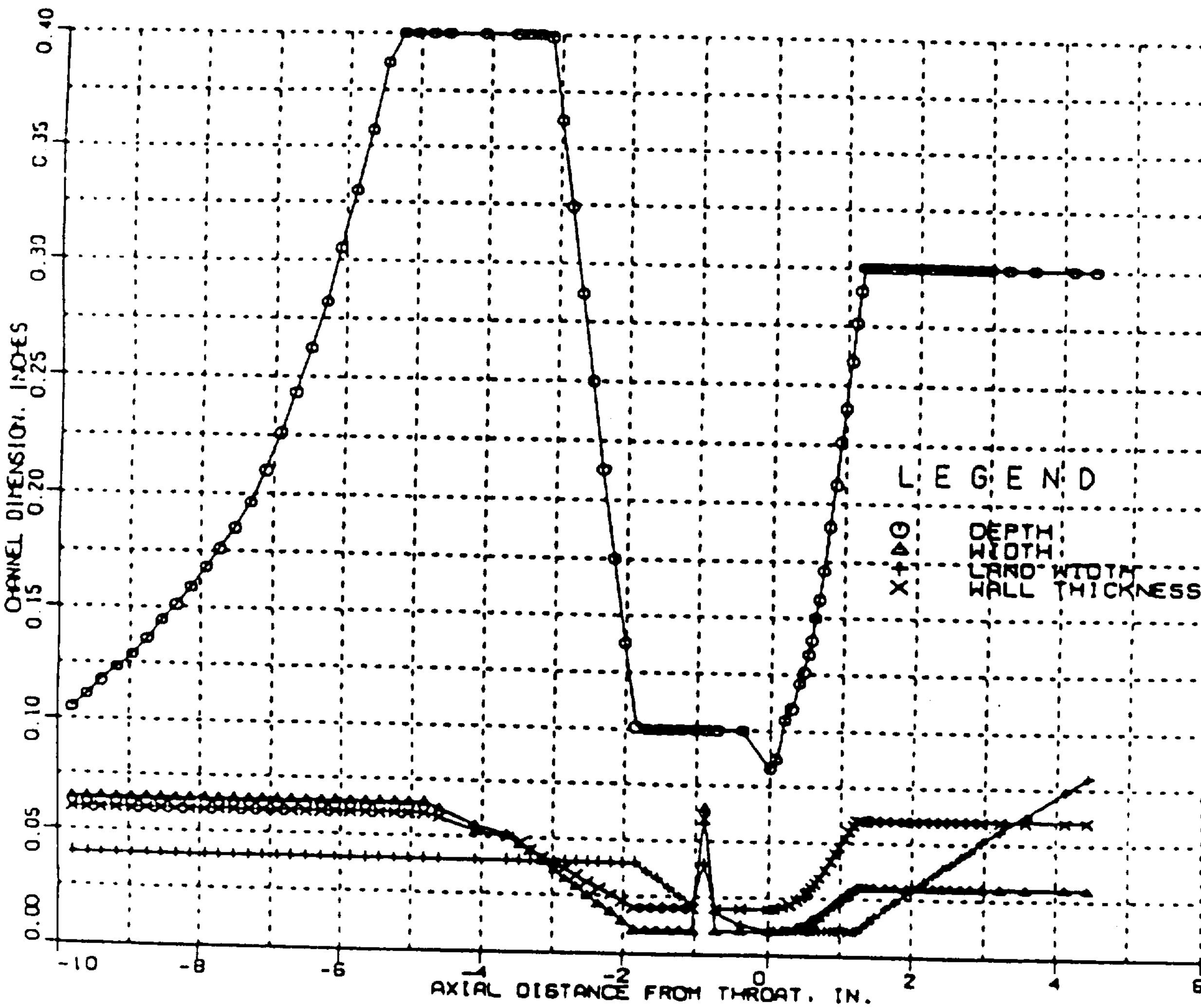


Figure 3.5.4-22 Regen Jacket Channel Dimension vs. Axial Distance from Throat

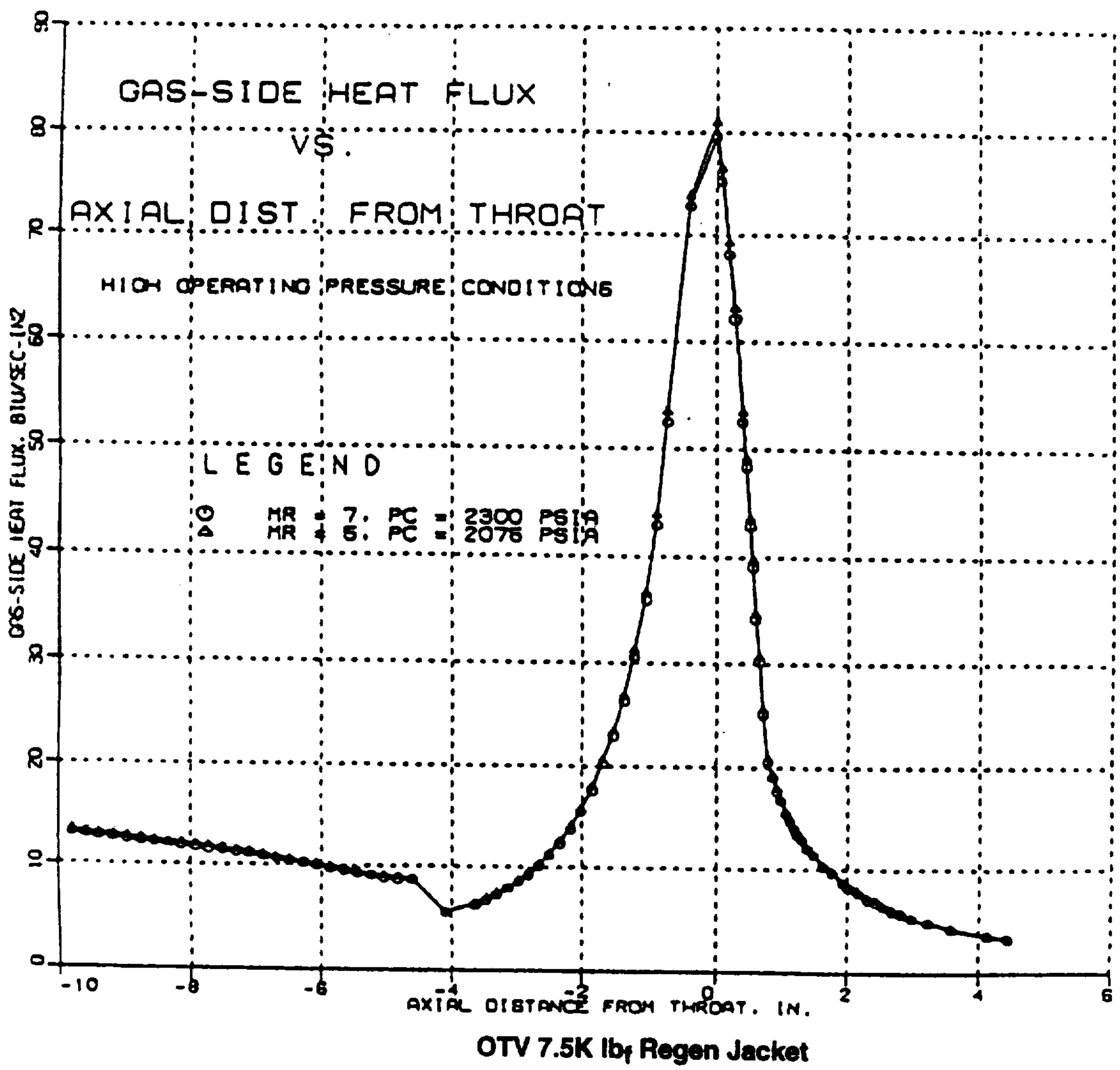


Figure 3.5.4-23 Gas Side Wall Heat Flux vs. Axial Distance From Throat at P_C >2000 psia

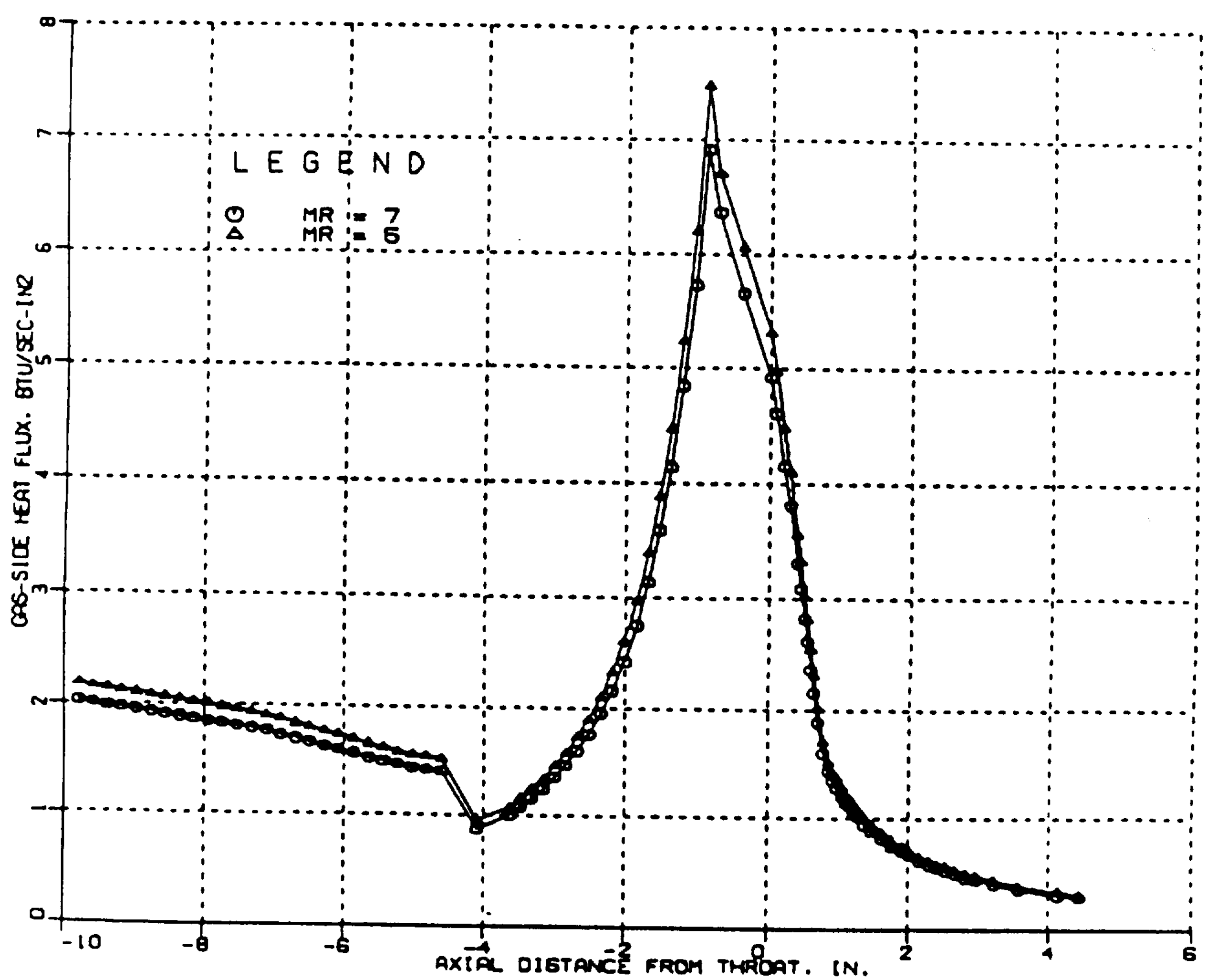


Figure 3.5.4-24 Gas Side Wall Heat Flux vs. Axial Distance From Throat at P_c = 200 psia

The correlated Cg profile is shown in Figure 3.5.4-25. Since the injector element type and chamber contour are the same for both the 3K lbf and the 7.5K lbf thrust level engines, the test data of the 3K lbf thrust level engine was used to establish the gas side boundary conditions for the higher thrust engine. The referenced test was believed to be an acceptable representation of heat flux trends for the OTV 7.5K lbf design due to the similarity in mixture ratio between the test data and the nominal MR in the final design. The MR in test # 147 was 6.2, whereas, the nominal MR for the 7.5K lbf engine is 6.0.

In test # 147 (Pc = 517 psia), the uncorrected Cg values describing the convergent section and throat heat flux conditions indicate that the flow relaminarized during acceleration. However, the relaminarization is not expected to occur when the Pc increases to 2000 psia at the high thrust operating condition due to the high Reynolds number. Therefore, the Cg results from the converging/nozzle region are not used for the 7.5K lbf thrust level boundary condition evaluation. The throat Cg assumed in the original 3.0K lbf thermal design, without accounting for the flow relaminarization, was 0.81; this value was maintained in the 7.5K lbf design as was the Cg profile in the nozzle region. For the convergent section the Cg's are assumed to vary linearly as a function of area ratio from the value at the aft end of the barrel section to the value at the throat. Figure 3.5.4-26 illustrates the predicted converging section and nozzle Cg profile.

3.5.4.2 Regenerative Cooled Baffles

3.5.4.2.1 Background

The high contraction ratio of the chamber makes incorporation of a heat exchanger, extending into the combustion chamber possible. A regen cooled baffle assembly provides additional surface area exposed to the hot combustion gases. The added hydrogen temperature increase is required for the 2000 psia chamber pressure baselined for the dual propellant expander cycle.

As the TCA design was scaled up in thrust level, use of oxygen to cool a center heat exchanger (or centerbody) was not technically feasible, due to the poor cooling qualities of oxygen. In addition, the required increase in heat input to the hydrogen would have to come from the chamber resulting in a longer L'. To maintain the balance between a high exit area ratio, maximum performance, an engine storage envelope of 60 inches, and a reasonable L', the burden of heating the remaining hydrogen was put on a heat exchanger, this is the regenerator.

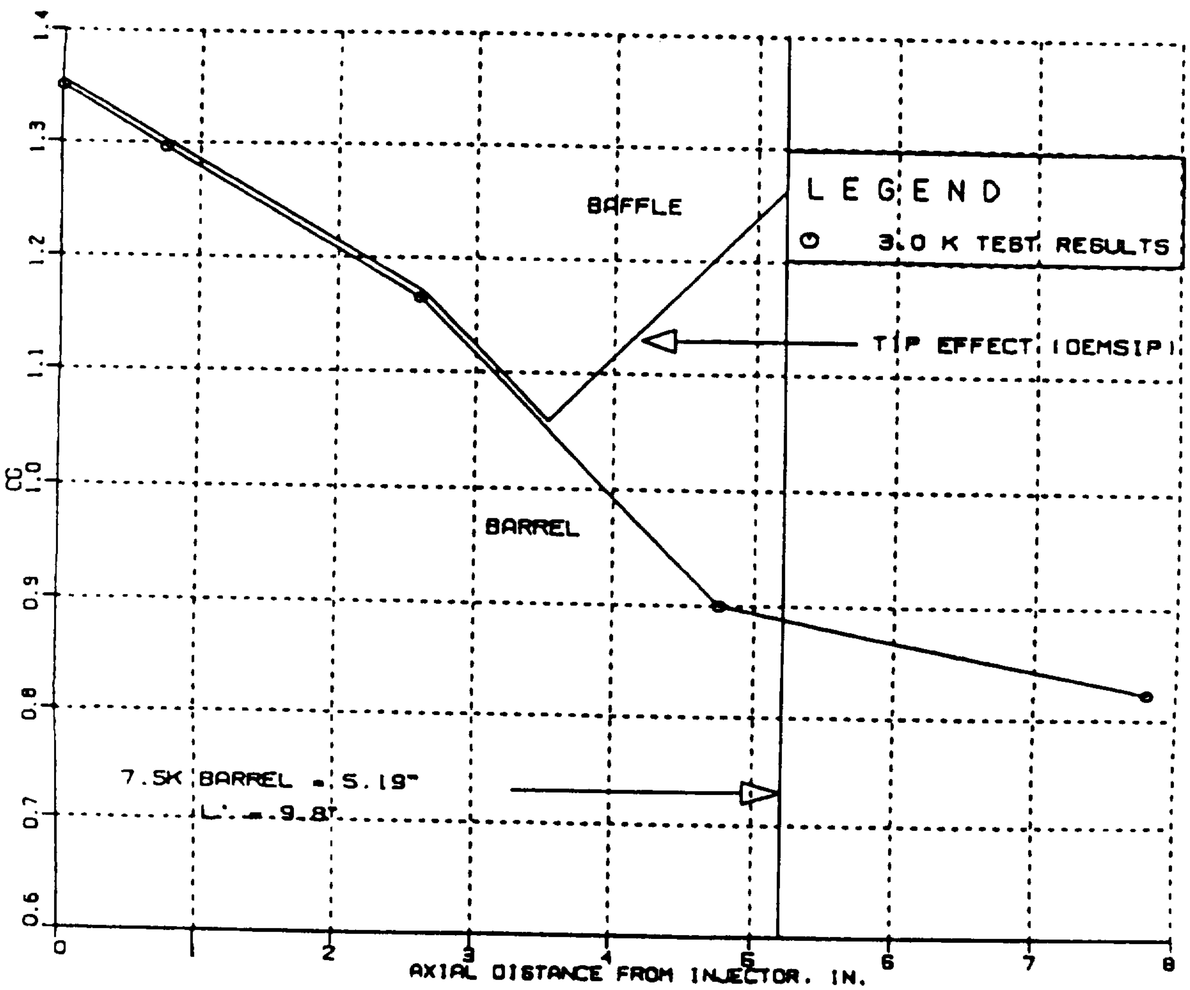
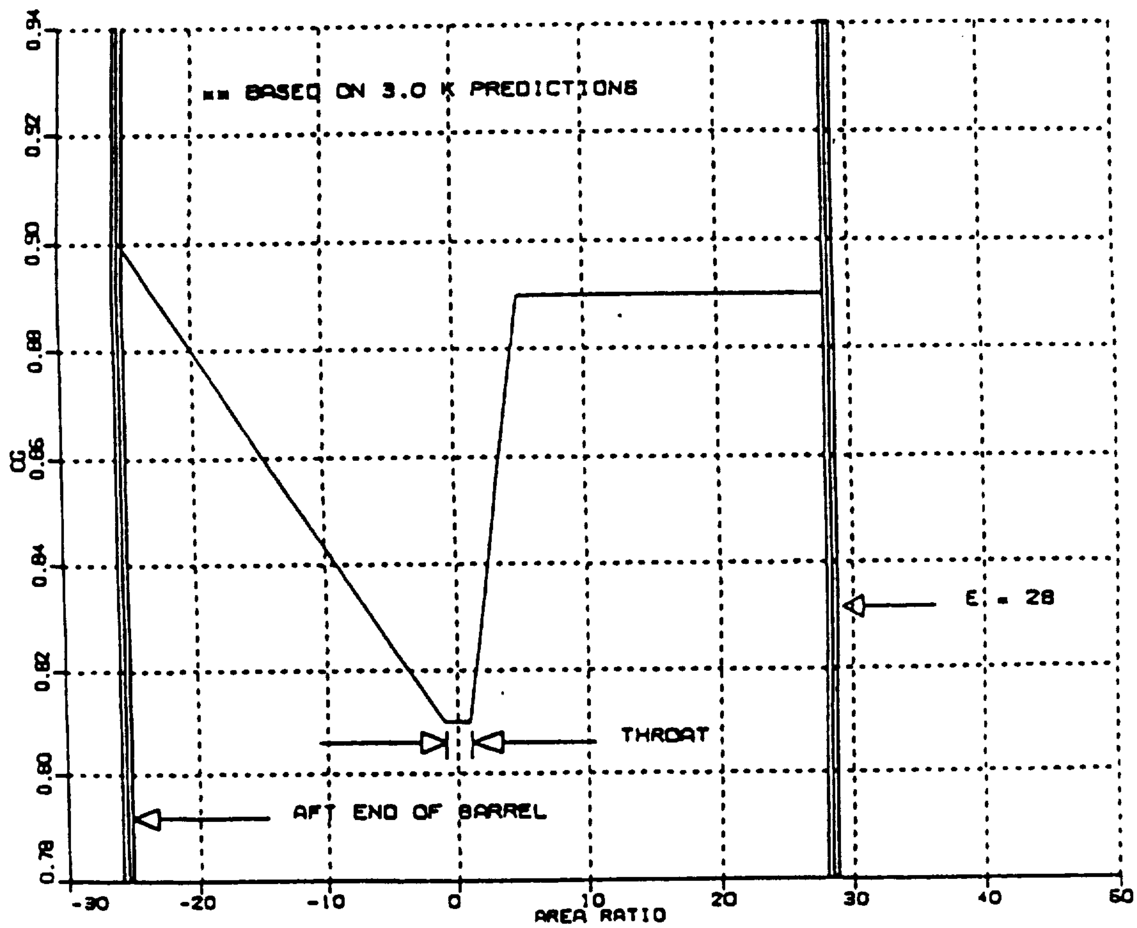


Figure 3.5.4-25 Regen Barrel/Baffle CG Profile vs. Axial Distance From Injector



OTV 7.5K lb_f Nozzle CG vs. Area Ratio

Figure 3.5.4-26 Regen Chamber CG Profile vs. Area Ratio

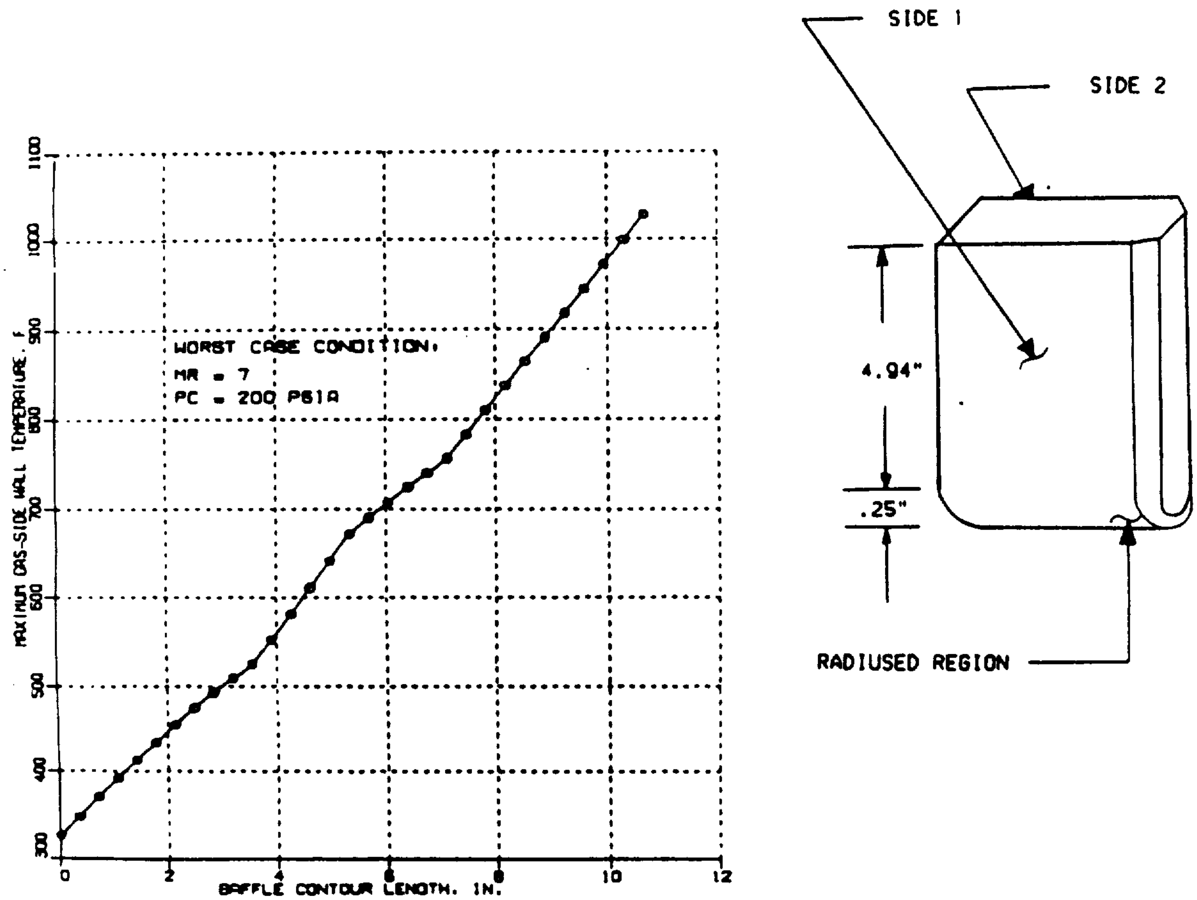
The initial design of the radial baffles included a center mounted igniter design. The igniter was flush with the injector face. During ignition, a hot combustion zone was present along the inner edges of the baffles. Subsequent thermal analysis of the baffle plates indicated that the corner channels, along the inner edge of the baffle, could not be cooled within the 1000°F wall temperature criteria. High temperatures were predicted during the low pressure operating conditions (Pc = 200 psia). The hydrogen velocity could not be increased to adequately cool the walls wthin design criteria without approaching sonic velocities.

Because of the severe thermal environment posed by the center mounted igniter, the igniter position was moved. By introducing the igniter through the side wall of the chamber, the ignition flame was not directed on the edges of the baffles. The edge of the baffles are difficult to cool because of the heat input from two sides. The faces of the baffles are easier to cool because heat is being input into the channel in one direction only. No further thermal analysis was undertaken at this time.

3.5.4.2.2 Analysis Conclusions

For baffle wall temperature control, the MR = 7, Pc = 200 psia operating condition represents the worst case. At MR = 7 the hydrogen flow rate is low and hence the cooling effectiveness is poor when compared to that at a mixture ratio of 5. Also, at the chamber pressure of 200 psia, the hydrogen bulk temperature at the exit of the baffle channels (which is assumed to be equal to the hydrogen inlet temperature at the turbine) is required to be approximately 820 degree F for power balance considerations (the corresponding required hydrogen exit temperature from the baffles for the Pc = 2300 psia, MR = 7, and Pc = 2075, and MR = 5 cases are only 540 degree F and 440 degree F, respectively). For the MR = 7, Pc = 200 psia condition a high hydrogen bulk temperature requirement coupled with a low hydrogen flow rate results in a poor combination of parameters in terms of controlling the maximum gas side wall temperature. At this operating condition the maximum baffle gas side wall temperature at the nominal flow rate and channel geometry is 1027 degree F. As previously noted, operating time below 500 psia chamber pressure would need to be restricted or the MR decreased to 4 or 5 to extend engine life.

Figure 3.5.4-27 illustrates the nominal gas side wall temperature profile for the baffle at the above mentioned operating condition. A benign boundary condition is assumed in the interior edge region adjacent to the igniter due to the hydrogen bleed in that area. At the baffle edge region adjacent to the chamber wall a gap of approximately 0.030 inches is assumed and is also considered to have a benign boundary condition. As a result, 48 of the 672 channels will not



TOTAL CONTOUR = SIDE 1 4.94" + RADIUS .25"R + SIDE 2 4.94" = 10.665"

Figure 3.5.4-27 Baffle Blade Maximum Gas Side Wall Temperature vs. Baffle Contour Length

see the same heat load as the remaining channels and are considered to have negligible heating. The maximum wall temperature occurs in the channel exit plane of the baffle when tolerance effects and a 4.0% flow maldistribution due to the 48 unheated channels are considered. The maximum predicted wall temperature is 1080 degree F for this condition.

A schematic of the baffle edge and side regions is shown in Figure 3.5.4-28. The dimensions for the edge channels are indicated in Figure 3.5.4-29. The corner channel in the baffle edge region adjacent to the igniter port is responsible for cooling a much greater surface area than its nearest neighbors. Several of the baffle edge concepts are shown in Figure 3.5.4-30. The heating is assumed to occur only in the radiused region at the aft end of the baffle (as described above) where the hydrogen bled from the igniter region mixes with the hot gas. A SINDA model of this channel based on its nominal dimensions was developed and a thermal analysis at a MR = 7, and Pc = 200 psia was conducted. The resulting maximum wall temperature as a function of local hydrogen temperature is shown in Figure 3.5.4-31. The anticipated local hydrogen temperature and resulting wall temperature is 155 degree F and 805 degree F, respectively.

The resulting regeneratively cooled baffle channel geometry is summarized in Table 3.5.4-5. There are a total of 672 channels in the baffle system. After the hydrogen exits the regeneratively cooled jacket near the injector face, it is plumbed to the baffles to provide cooling. The cooling channels in the baffle are a two pass design; the hydrogen flows down one side of each baffle (parallel to the gas flow), changes direction in the radiused region at the aft end of each baffle (see Figure 3.5.4-32) and counter flows back to its exit near the injector face.

3.5.4.2.3 Analysis Methodology

The baffle cooling channel geometry is designed to accommodate the conditions at the MR = 7, Pc = 200 psia operating condition. Because the required hydrogen bulk temperature at the baffle exit is high (near 820 degree F) for power balance considerations and the hydrogen flow rate is low due to the high mixture ratio, controlling the wall temperature is most difficult at this operating condition when compared to other conditions evaluated. The channels were designed assuming construction using a diffusion brazed platelet stack method. For ease of fabrication the channel geometry was designed to remain non-varying throughout the entire hydrogen flow path. As a result the gas side wall temperature is the lowest at the hydrogen inlet to the baffle where the hydrogen is the coolest and increases as the hydrogen heats up. The minimum wall thickness is determined in the same manner as the regen cooled channels. The number of

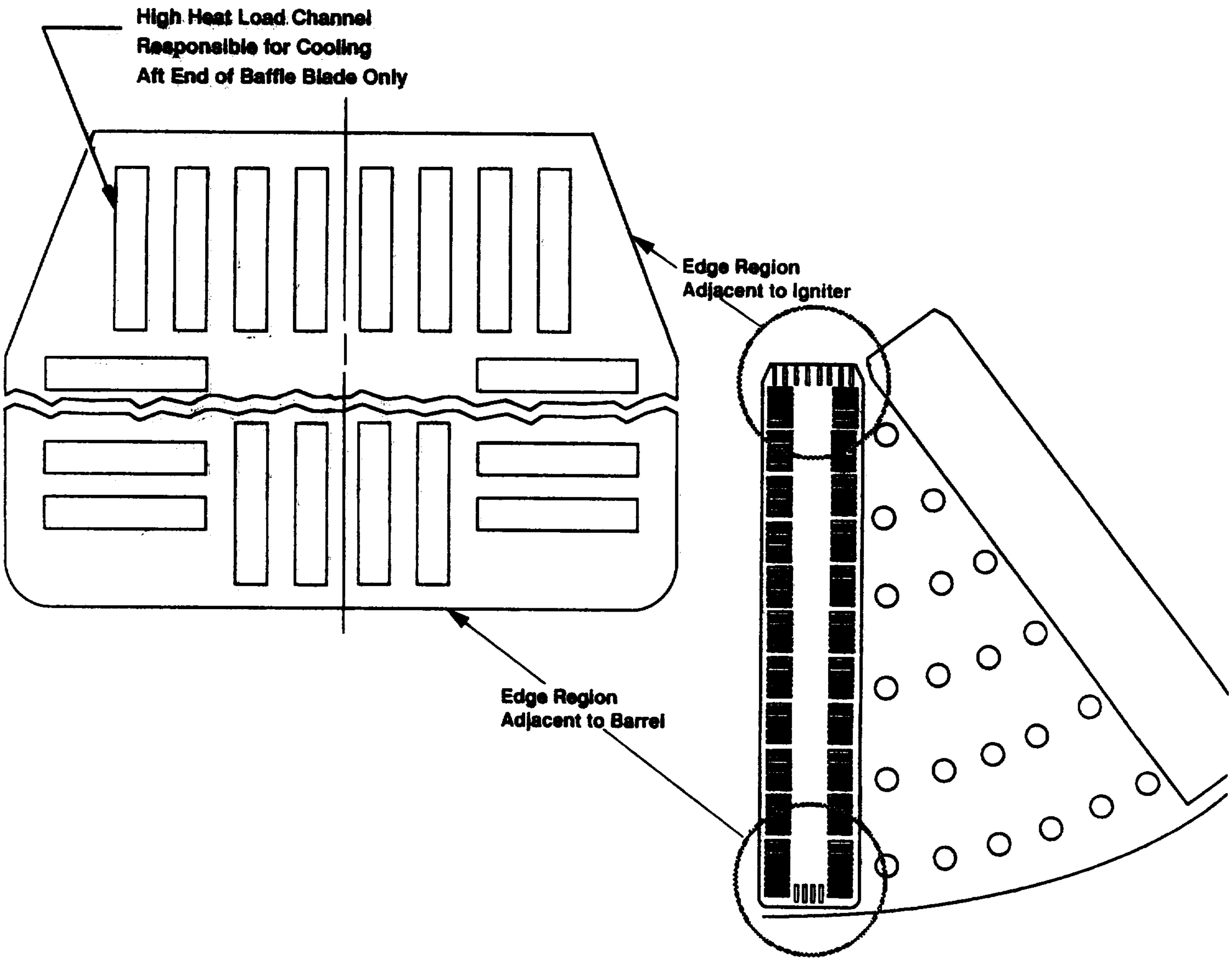


Figure 3.5.4-28 Baffle Schematic

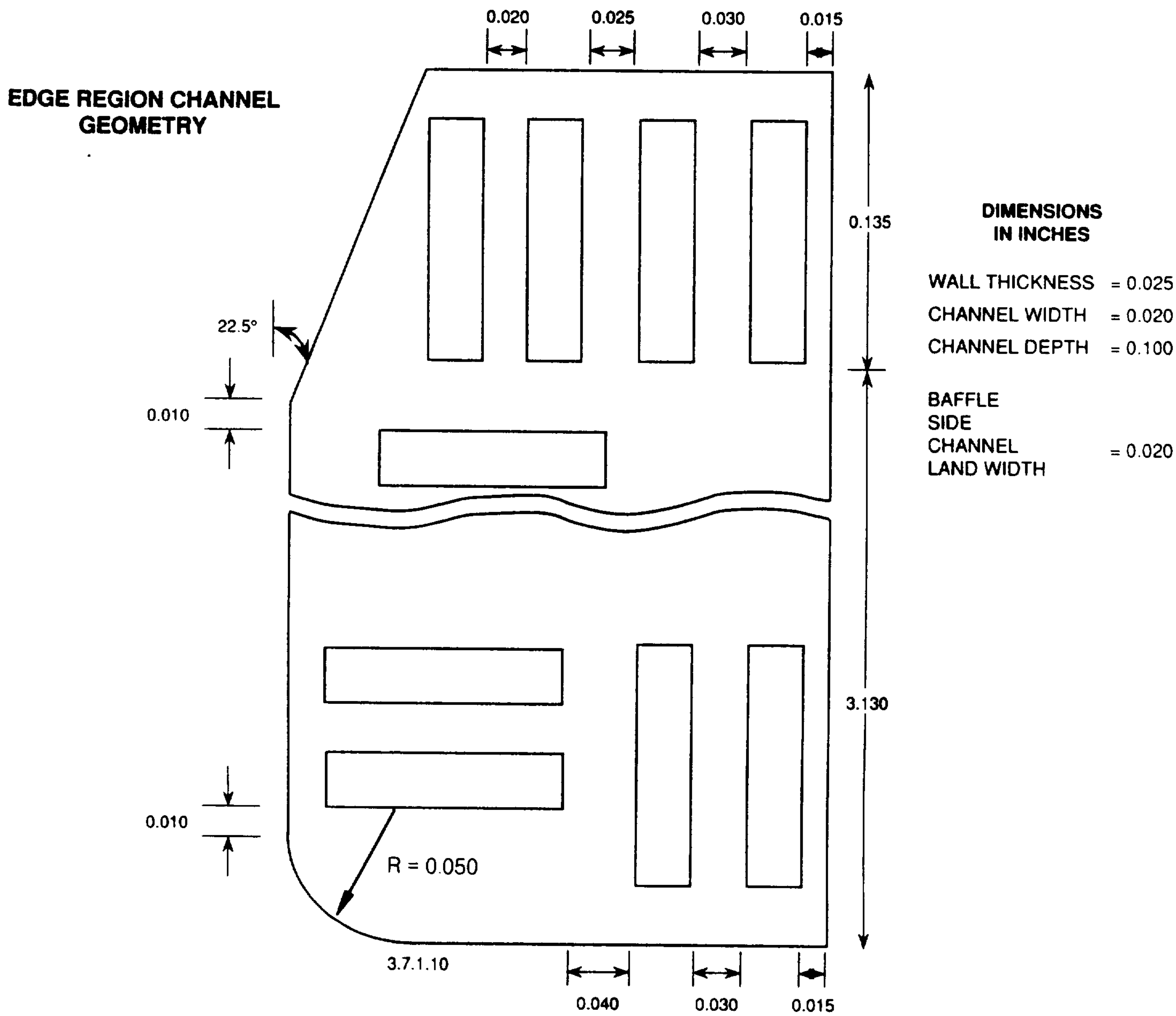


Figure 3.5.4-29 Baffle Edge Region Channel Geometry

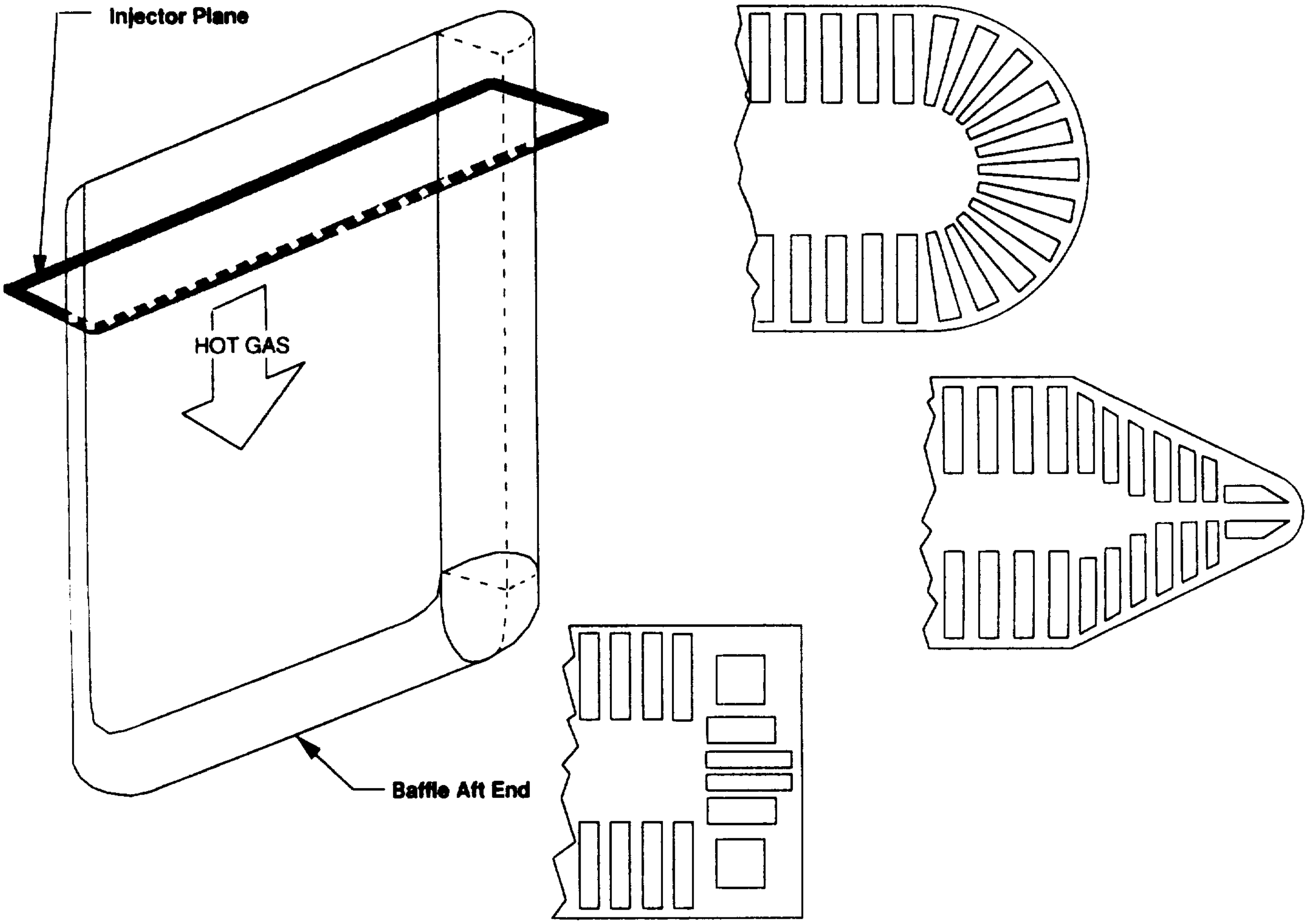


Figure 3.5.4-30 Baffle Edge Concepts

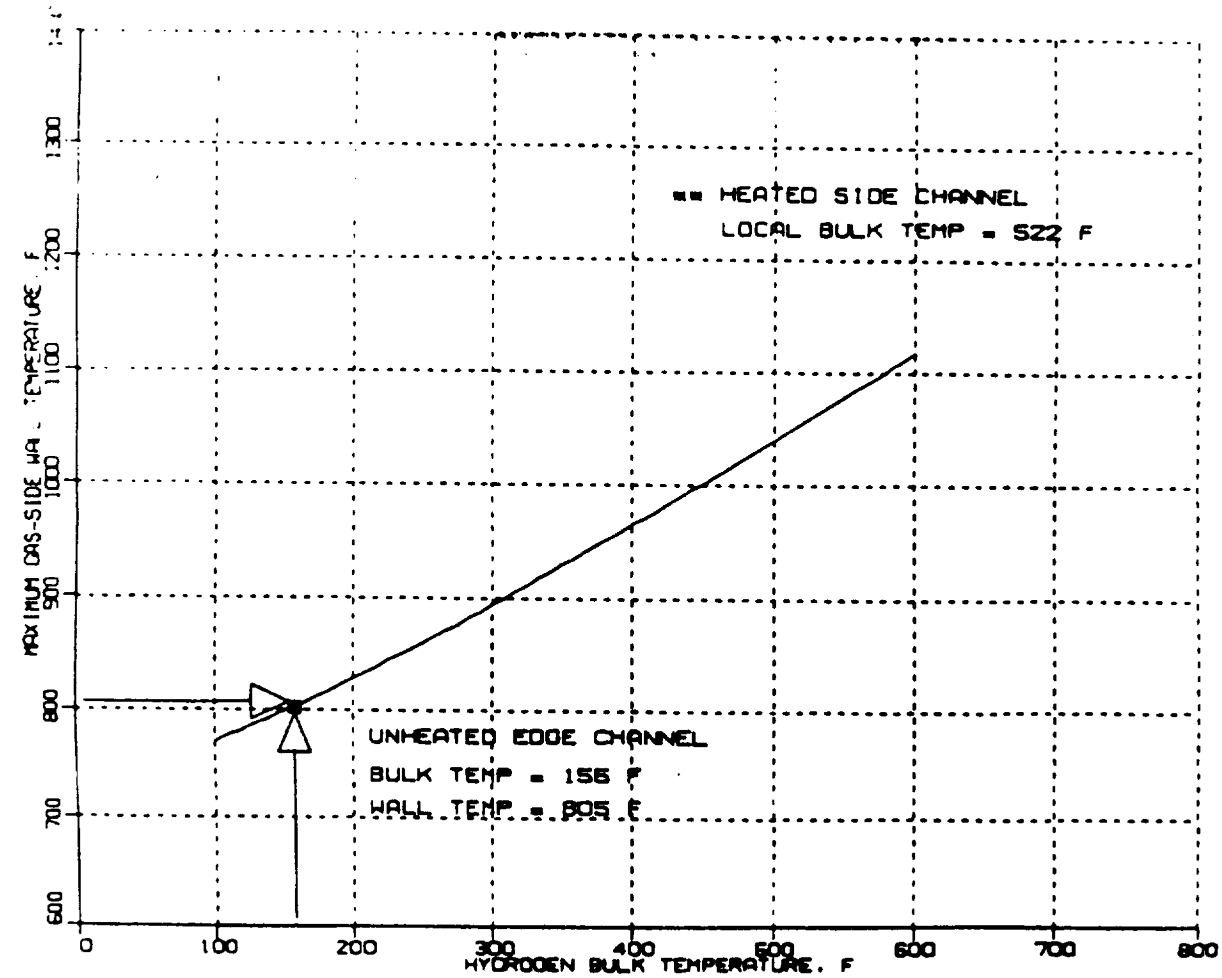
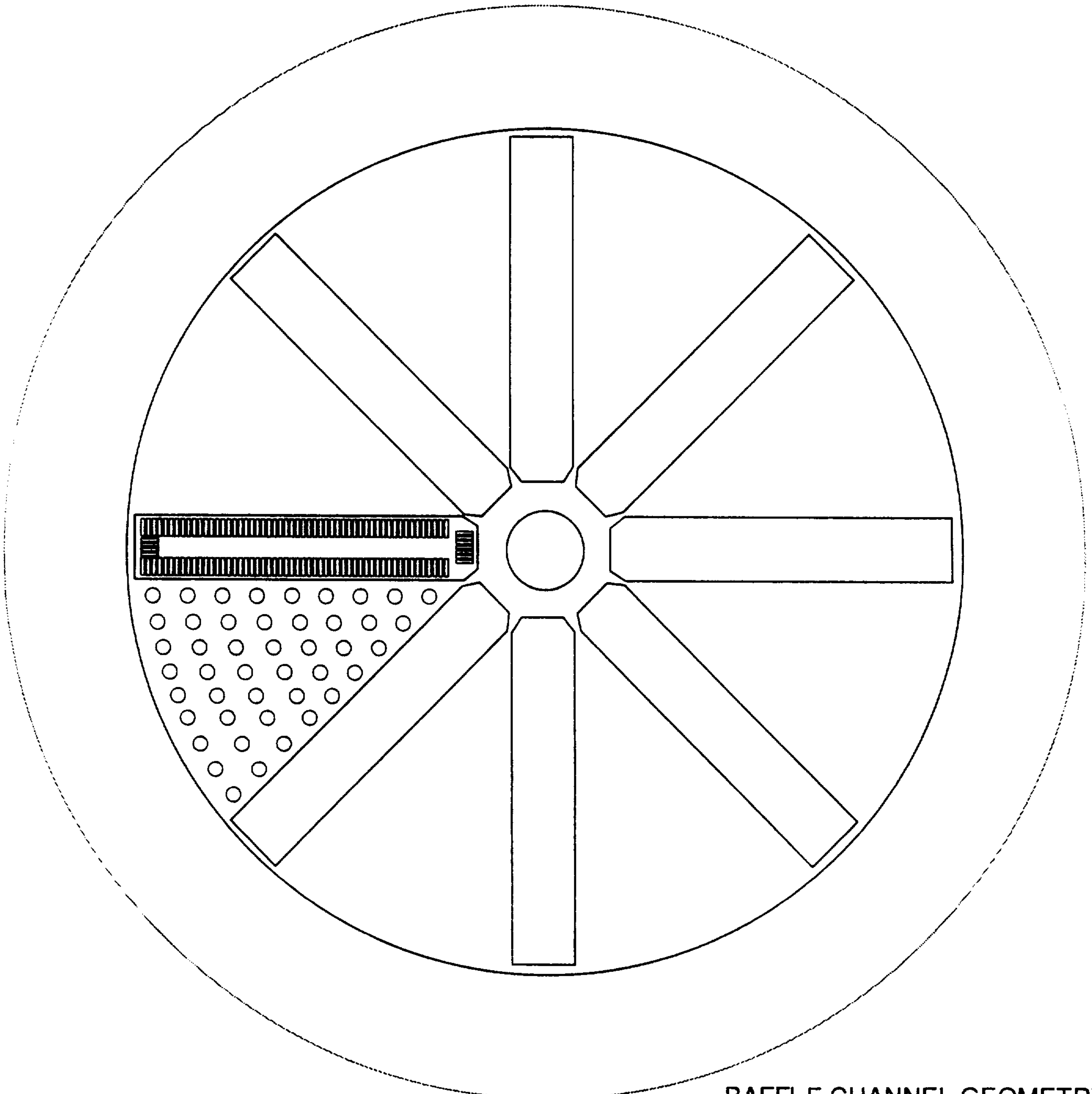


Figure 3.5.4-31 Baffle Unheated Edge Channel Maximum Gas Side Wall Temperature vs. Hydrogen Bulk Temperature

Table 3.5.4-5

Summary - Regen Baffle Channel Geometry

ORBITAL TRANSFER VEHICLE ENGINE INJECTOR DESIGN
BAFFLE ARRANGEMENT



BAFFLE CHANNEL GEOMETRY

No. Chans	672
Dept (In.)	0.100
Width (In.)	0.020
Land (In.)	0.020
Wall Thickness (In.)	0.025

OTV BAFFLE BLADE SCHEMATIC

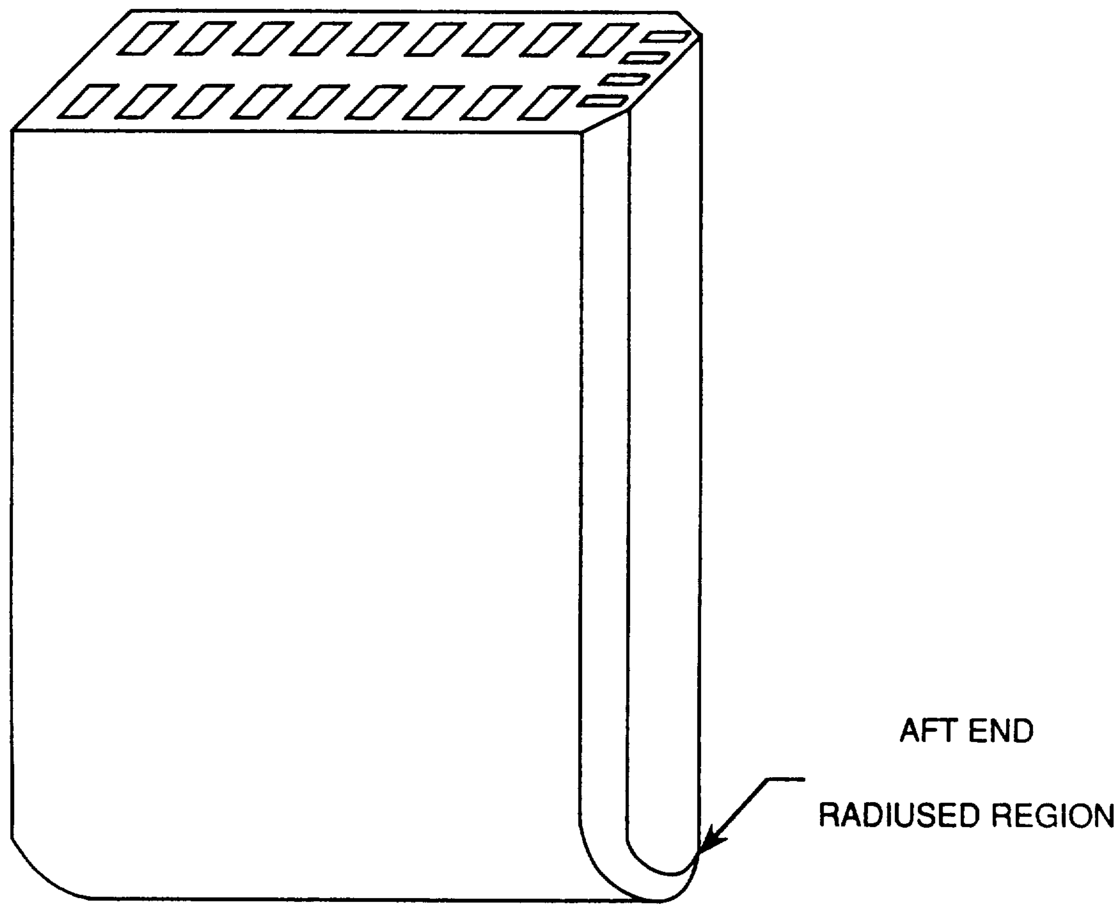


Figure 3.5.4-32 Baffle Blade Schematic

cooling channels and flow rate through each channel is determined based on the total exposed surface area of the baffle for a given land and channel width.

The baffle blade local Cg's used in the 7.5K lbf baffle design are based on the 3K lbf outerbody Cg's coupled with heating trends inferred from the GEMSIP Baffle Heat Transfer Analysis (Reference 3).

The GEMSIP baffle blade study indicates that there is an unusually high heat transfer rate at the end of the baffle (referred to as the "tip effect"). This enhancement was believed to be caused by either a zone of high combustion turbulence or a boundary layer separation. The first explanation may be a function of injector type and combustion characteristics and thus would be difficult to quantify without more data. The latter explanation would describe the tip effects as a function of the flow characteristics in the vicinity of the blade tip and would be independent of blade length. In order to provide a better understanding of the cause of the enhancement, a study using different lengths of baffle blades would be needed, however this type of study is beyond the scope of work under this current contract. Therefore for OTV boundary condition evaluation the heat transfer behavior near the blade tip is treated as if the phenomena is due to flow separation caused by the physical presence of the baffle tip.

The GEMSIP results are illustrated in Figure 3.5.4-33. The flux begins to increase dramatically as the tip is approached from the side value corresponding to an axial distance 1.6 inches upstream of the tip (the steady increase in heat flux from the injector to this side point shown in Figure 3.5.4-33 is attributable to the entrainment of hot gas into GEMSIP's film cooling region). The tip flux increases to approximately 120% of this side value. The local Cg value for the OTV 7.5K lbf engine at the blade tip is assumed to follow the same trend. The resulting baffle Cg profile is included in Figure 3.5.4-34.

3.5.4.3 Heat Transfer Correlations

3.5.4.3.1 Coolant Side Heat Transfer Correlations

The coolant side heat transfer and pressure drop correlations utilized are the same for the regeneratively cooled chamber and baffles. A brief description follows.

The forced convention heat transfer coefficients for hydrogen are evaluated using the Hess and Kunz correlation (Reference 4).:

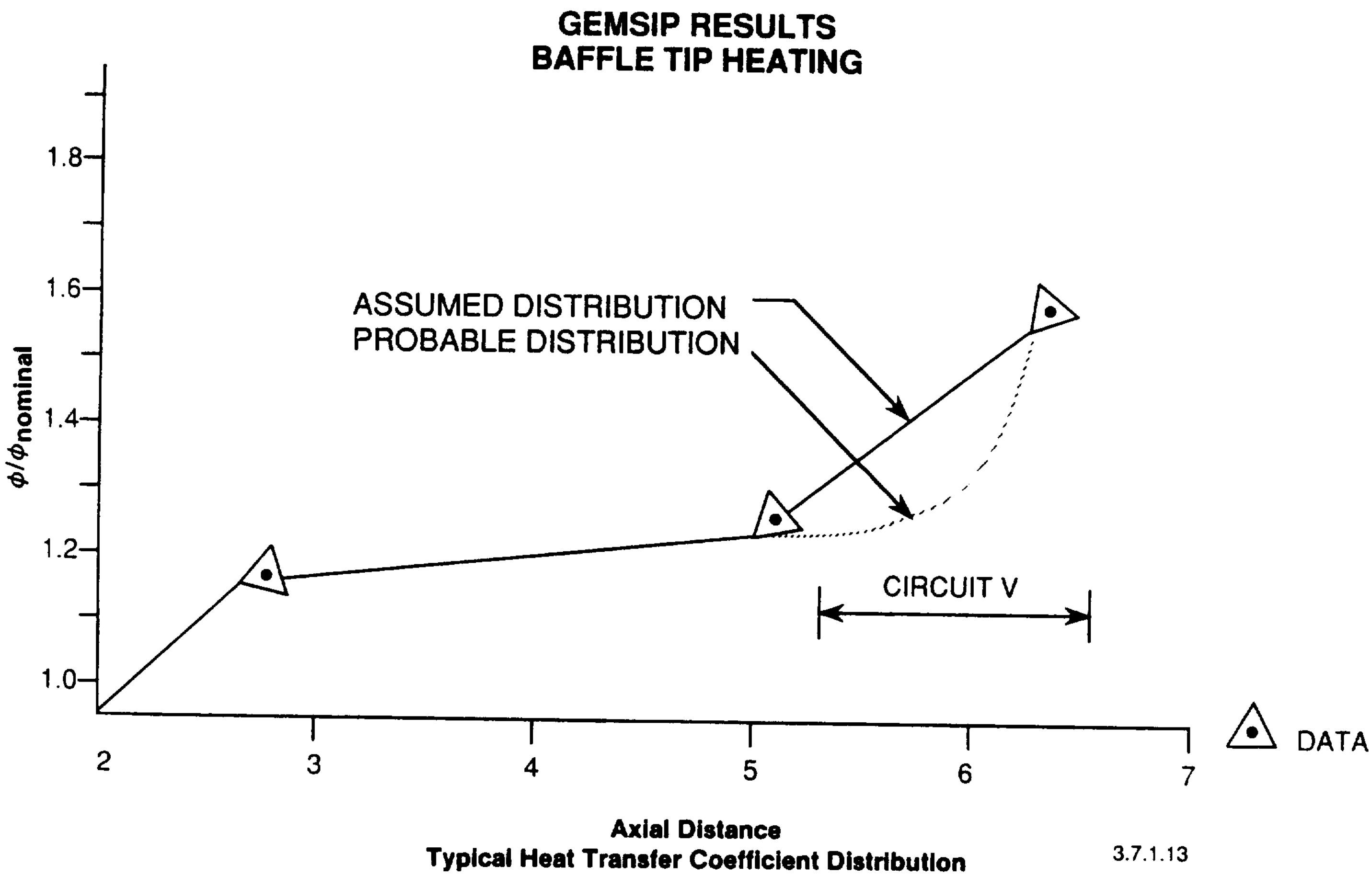


Figure 3.5.4-33 GEMSIP Results - Baffle Tip Heating

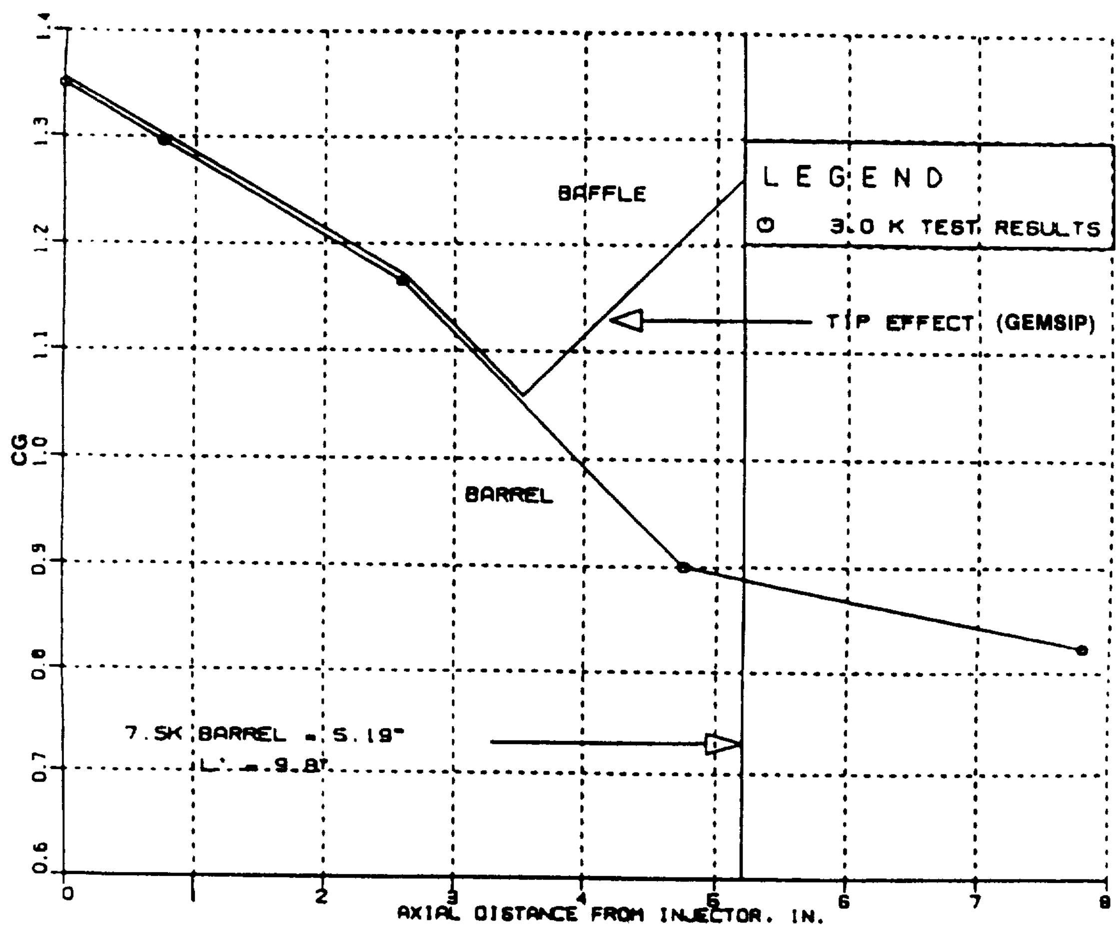


Figure 3.5.4-34 Barrel/Baffle CG vs. Axial Distance from Injector

$$Nu_f = .0208\, Re_f^{0.8}\, Pr_f^{0.4} (1. + .01457 \frac{\mu_w \rho_b}{\mu_b \rho_w}$$

where:

Nu_f	=	Nusselt Number
Re_f	=	Reynolds Number
Pr_f	=	Prandtel Number
μ_w	=	fluid viscosity, wall temp
μ_b	=	fluid viscosity, bulk temp
ρ_w	=	fluid density, wall temp
ρ_B	=	fluid density, bulk temp

The boundary layer in the channel is assumed to be fully developed at all locations in the baffle and regen-cooled jacket.

3.5.4.3.2 Coolant Pressure Drop

The coolant pressure drop within a cooling channel relating the exit static pressure to the inlet stagnation pressure is calculated as:

$$\Delta P_s = \Delta P_{inlet} + \Delta P_{fric} + \Delta P_{dyn}$$

The inlet pressure loss was calculated assuming a 1/2 dynamic head loss for flow contraction. The friction pressure loss was based on the following friction factor:

$$f = .0055\, [1. + (2.0E4 * \varepsilon / D + 10^6/Re)^{.33333}]$$

where:

f	=	friction factor
ε	=	wall roughness
D	=	hydraulic diameter
Re	=	Reynolds number

A roughness of 125.0 E x10^{-6} inches was used. Baffle turn losses associated with the radiused region of the baffle were negligible. These losses (Figure 3.5.4-35) for the

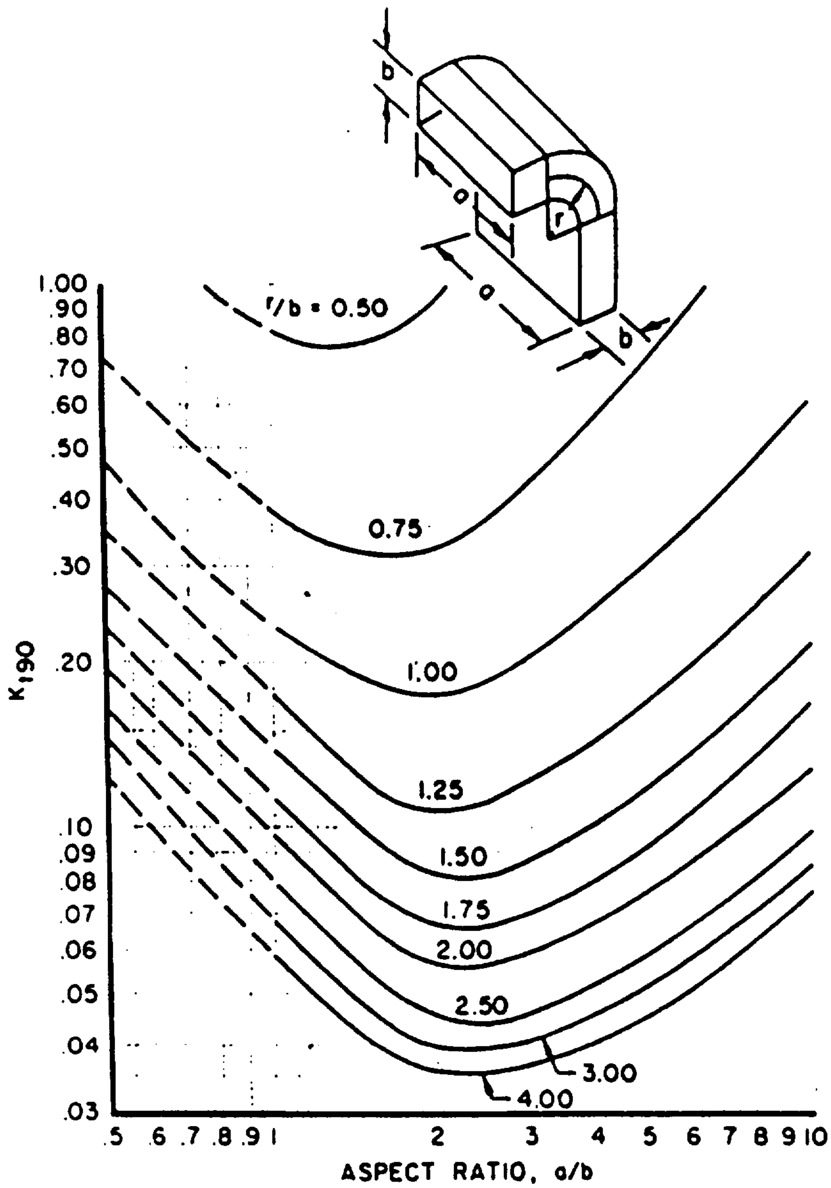


Figure 3.5.4-35 Aspect Ratio vs. Flow Losses

heated channels were estimated from resistances of a rectangular duct and found to have a dynamic head loss of less than 0.03. Because the dynamic head loss is low, these losses are not included in the pressure loss calculation.

3.5.5 Mechanical Design

This section describes the design of the test bed Thrust Chamber Assembly (TCA) for the 7.5K lbf thrust level OTV engine. The test bed TCA components, illustrated in Figure 3.5.5-1, include: igniter assembly, injector, baffles, and hydrogen cooled outer chamber and throat. Modifications for the flight weight TCA are based on this design and discussed elsewhere in this report.

3.5.5.1 Thrust Chamber Assembly (TCA)

The baseline 7.5K lbf thrust level TCA test bed conceptual design is shown in Figure 3.5.5-1. The igniter is mounted on the side, however, rather than at the center as assumed in the heat transfer analysis. This will affect the boundary conditions for the baffle plate edges and the final design. Removal of the igniter from the center provides a more benign thermal environment for the baffles during ignition.

The chamber was designed to use NASA-Z copper with a electroformed NiCo alloy closeout (see Reference 5). The inlet for the hydrogen coolant is at the aft nozzle at a 28:1 exit area ratio. Hydrogen coolant flows axially counter to the combustion gas flow. Design of the coolant channel contour was discussed in Section 3.5.4-1. The channel geometry shown reflects the lowest pressure drop configuration obtainable given the coolant flow rate and requirements. The chamber is bolted to the injector with 16 high strength bolts. In a flight chamber this would be a welded assembly.

Manifolding to route the hydrogen from the regen chamber to the baffle plate inlet is contained within the flanges. The chamber coolant channels flow into a common annular manifold. Localized ports deliver the high pressure hydrogen to each of the 8 baffle plates. The two pass hydrogen cooled baffle exits into a tubular manifold which is then connected to the turbine inlet. The baffle plates are individually formed from a diffusion bonded ZrCu platelet stack.

A center post containing active injector elements balances the void left by the encompassing baffle plates. This center post extends the length of the baffle, thus the baffle is

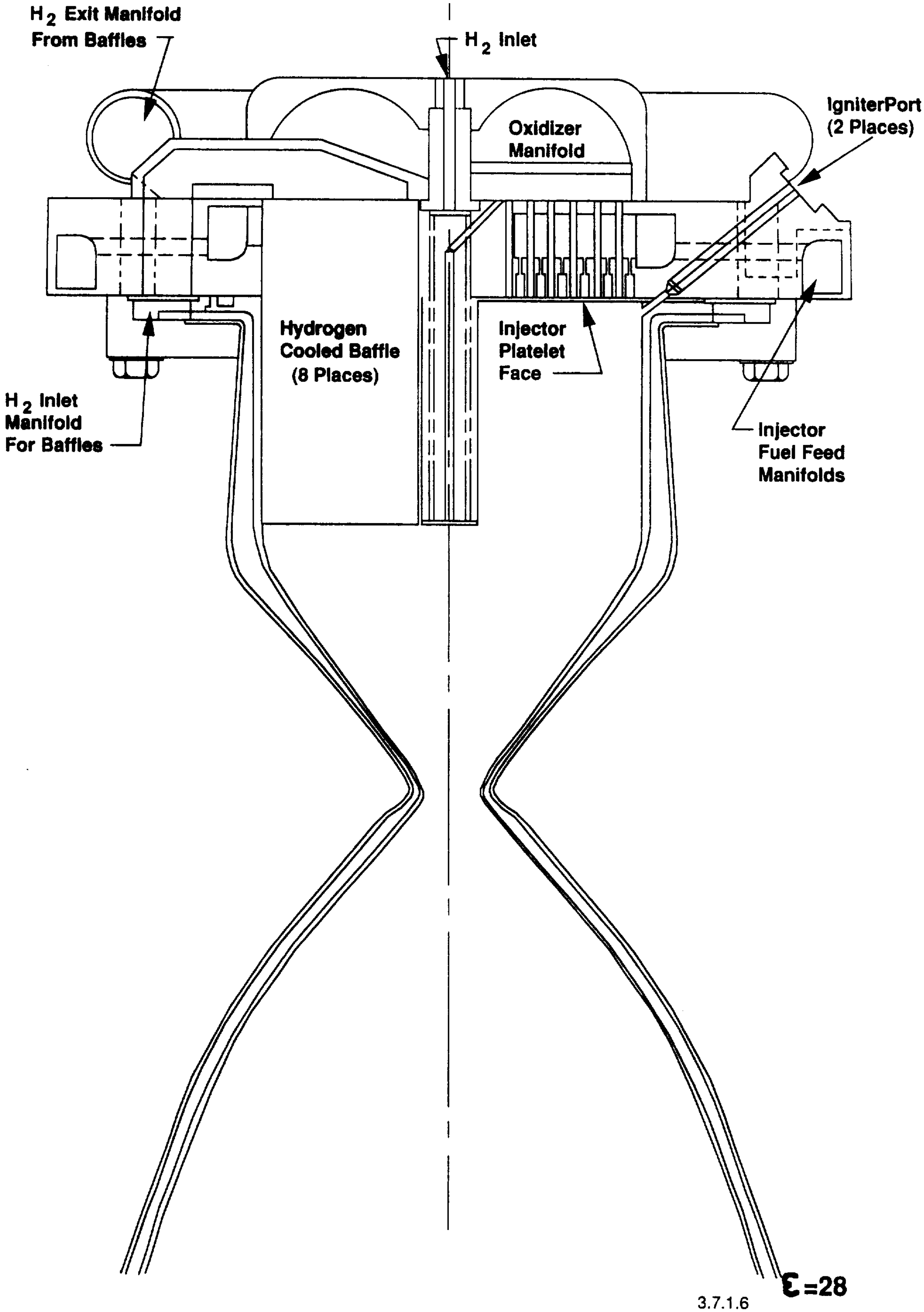


Figure 3.5.5-1 7.5K lbf OTV Thrust Chamber Assembly Concept

protected against heat input along the edges. The edge of the baffle is the most difficult to cool since there is heat input from two sides. Hydrogen is separately routed through the middle to feed the elements (Figure 3.5.5-2). The oxidizer flow is diverted internally from the oxidizer manifold. Analysis of this component has not been completed to determine number and size of elements, and coolant passage size.

The hydrogen is delivered to the injector though the side into an annular manifold. Cross drilled holes connect to a center fuel feed annulus where EDM'd slots feed each injector element. The oxidizer dome, welded to the backside of the injector, delivers the oxidizer to the through-drilled injector passages.

A diffusion bonded Nickel face plate forms the "I" triplet element. The possibility of using the high strength nickel base Inconel alloys for the injector manifold is being considered to reduce weight.

3.5.5.1.1 Igniter

The ignition mechanism, also referred to as a "torch igniter" consists of a aviation type spark plug with separately plumbed fuel and oxidizer lines providing an oxidizer rich flame through the core (Figure 3.5.5-3). The igniter assembly is introduced through the side of the injector. Two assemblies are used in accordance with the manrating requirement.

This igniter design was verified during hot fire tests using the 3K lbf test hardware. The actual hardware is shown in Figure 3.5.5-4.

3.5.5.1.2 Injector

The injector design is based on the "I" triplet premix element (Figure 3.5.5-5), tested as part of the 3K lbf thrust level program. This injector consisted of 2 annular rows of elements (36 elements per row) canted at 20 degrees around the centerbody. Thrust per element rating for the 3K lbf injector was 45 lbf for the 36 elements in the outer row and 38 lbf for the 36 elements in the inner row. Baseline element design for the 7.5K lbf injector utilized the geometry of the outer row of elements on the 3K lbf design. Utilization of an 8-baffle blade geometry for the 7.5K lbf TCA design divides the injector into 8 pie shaped segments. The thrust rating per element yielded 21 elements per segment for a total of 168 elements. Table 3.5.5-1 offers a comparison of the baseline designs for the 2 thrust level TCA's. Figure 3.5.5-6 illustrates the injector designs discussed.

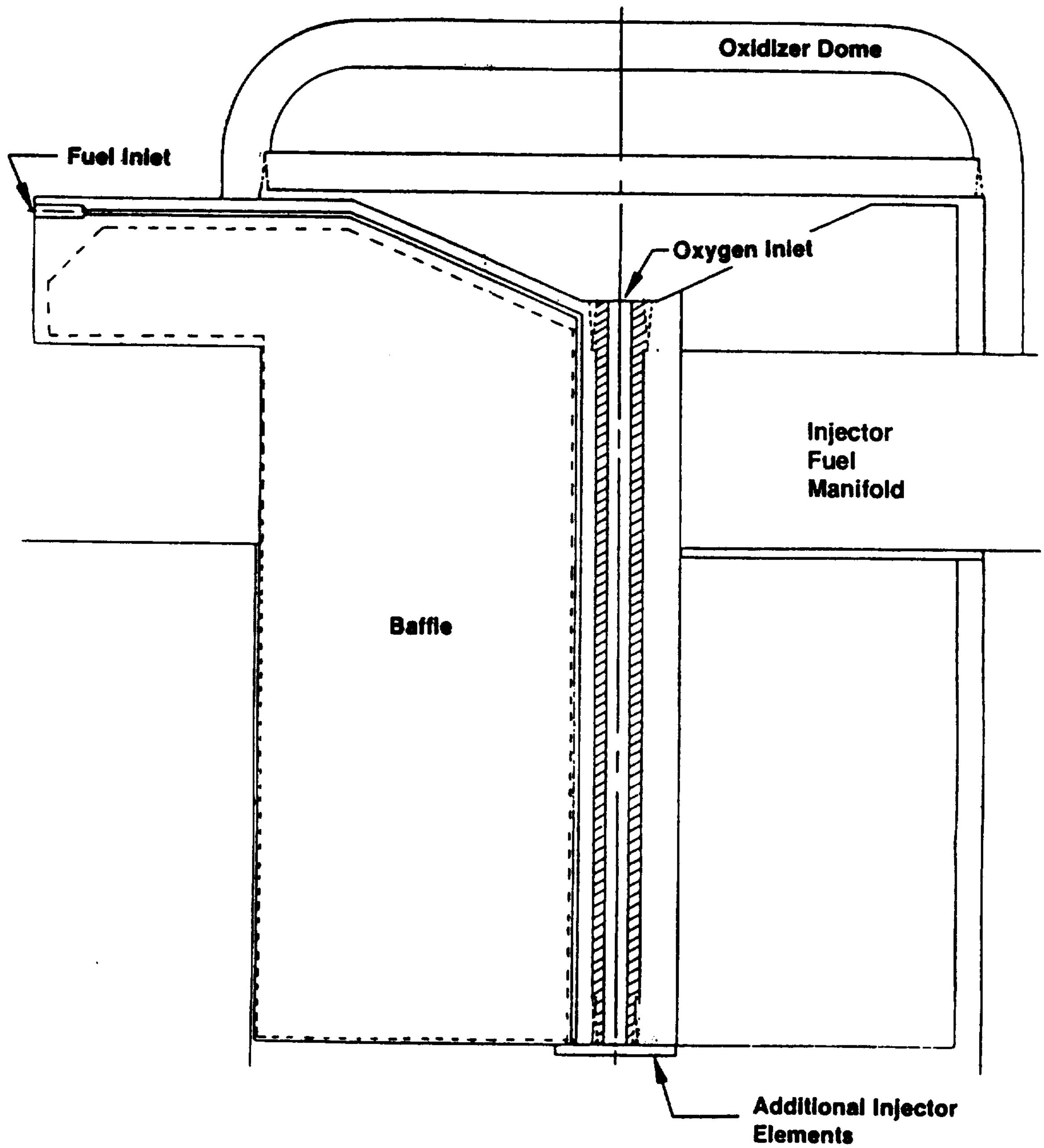


Figure 3.5.5-2 Center Mounted Injector Elements

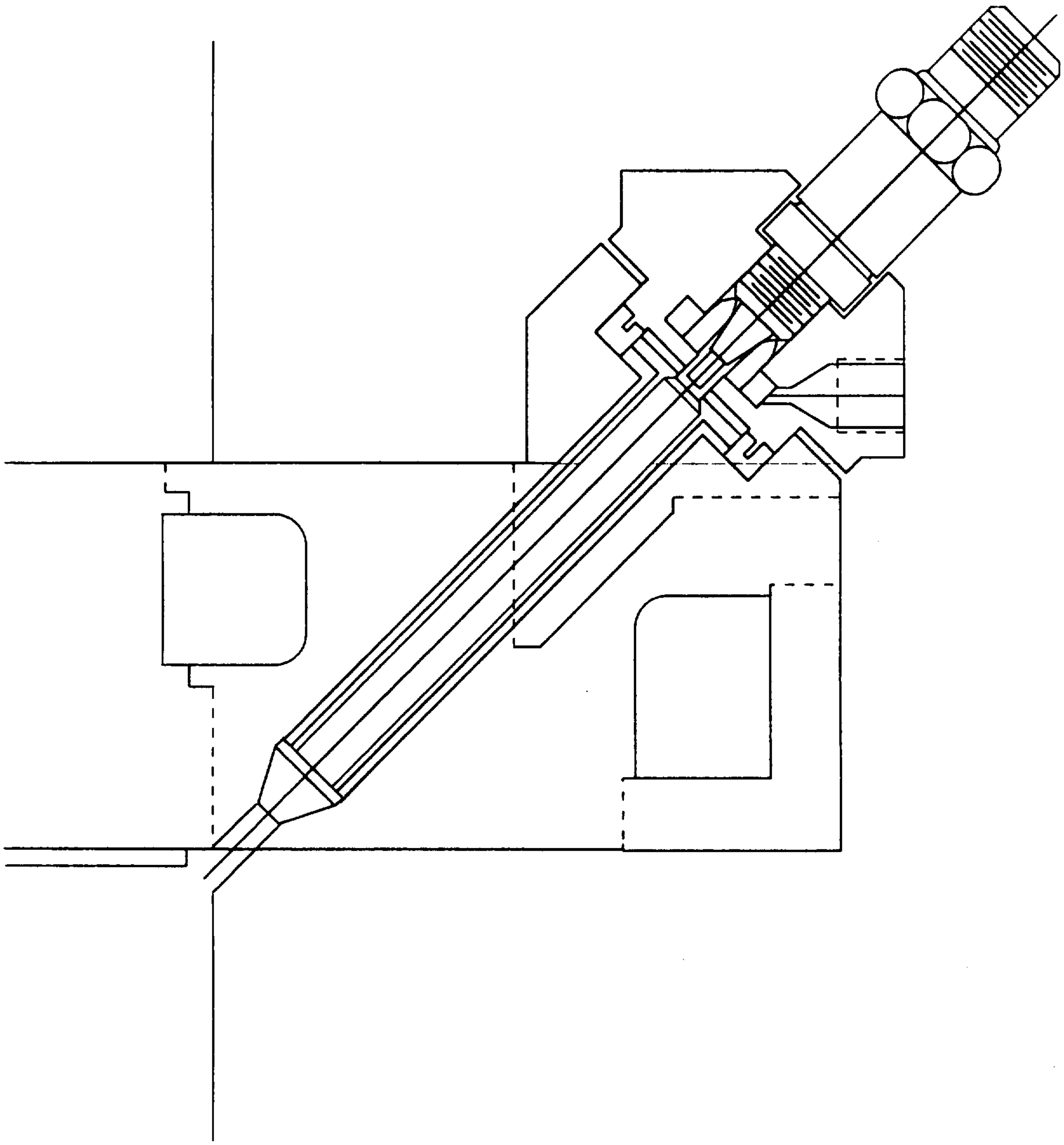


Figure 3.5.5-3 7.5K lbf OTV Torch Igniter

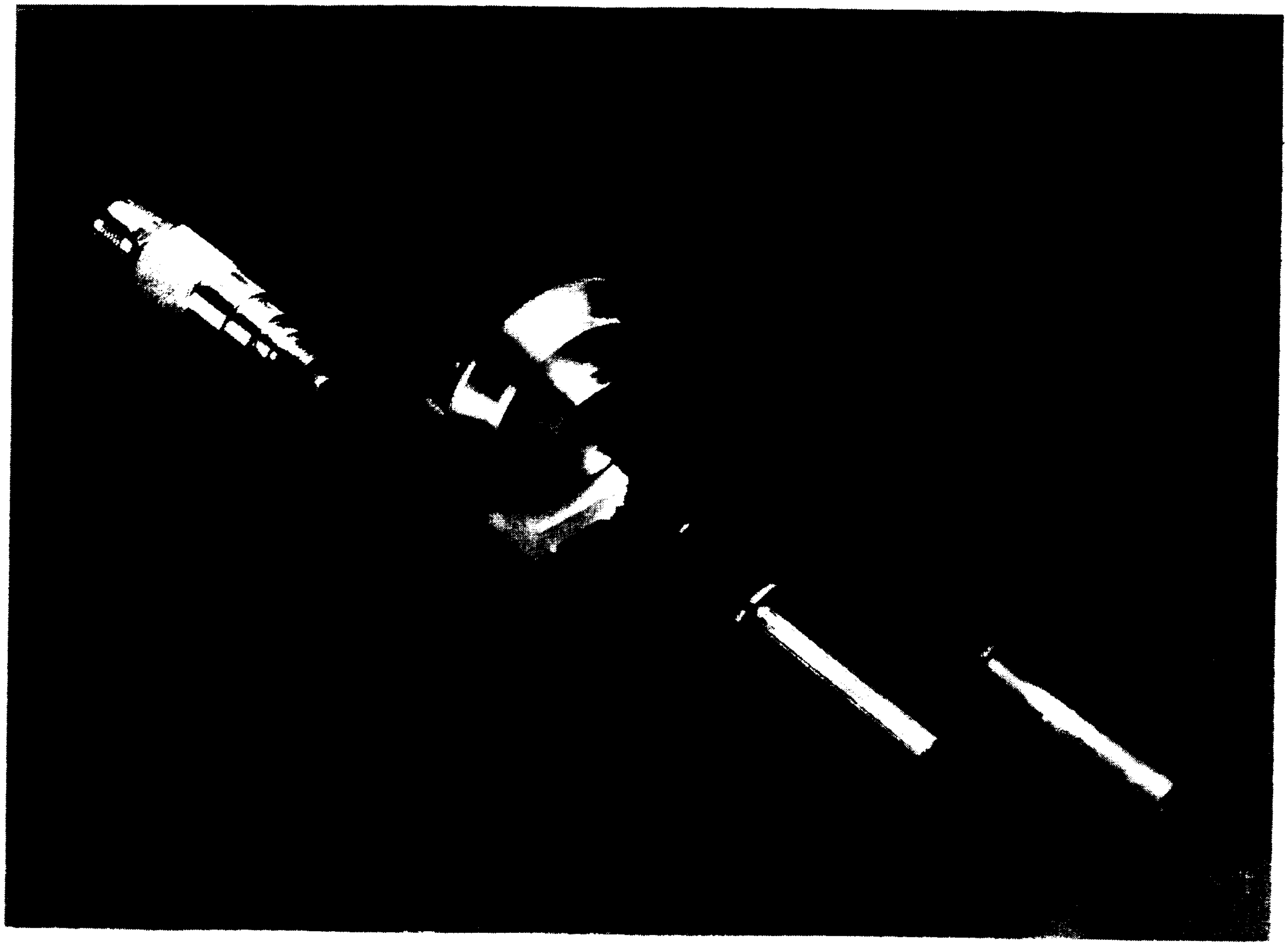


Figure 3.5.5-4 Disassembled Torch Igniter

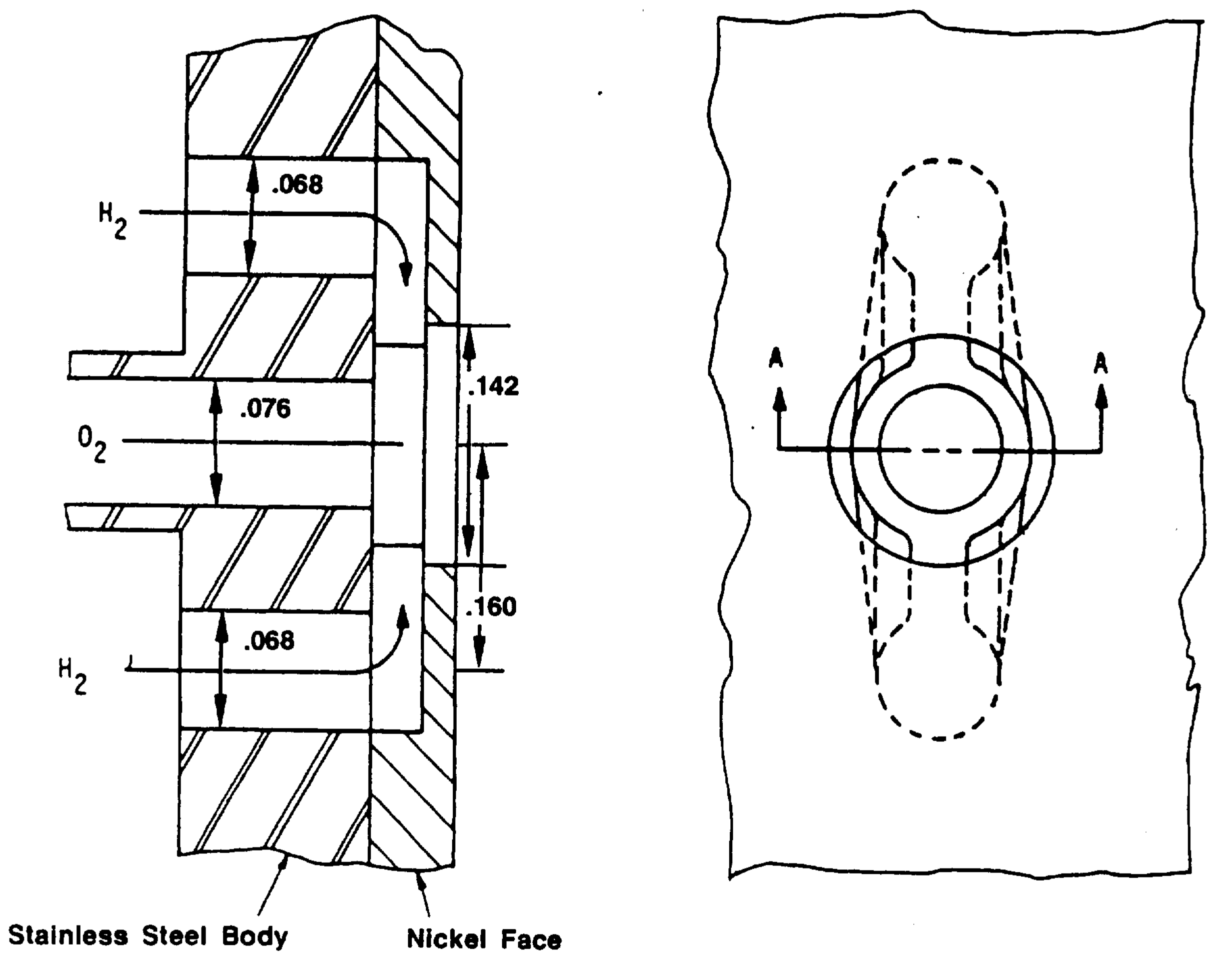


Figure 3.5.5-5 Modified "I" Triplet Premix Injector Element

Table 3.5.5-1
Uprated OTV Thrust Chamber Design Concept

DESIGN ELEMENTS	3K lbf	7.5K lbf
CHAMBER PRESSURE, Pc (psia)	2000	2000
MIXTURE RATIO, O/F, NOMINAL	6.0	6.0
COMBUSTION CHAMBER DIAMETER (in.):		
O.D.	5.62	7.64
I.D.	4.00	-
COMBUSTION CHAMBER LENGTH (in.)	15.0	11.25
NOZZLE THROAT DIAMETER (in.)	0.960	1.518
CONTRACTION RATIO	16	>17
COMBUSTION STABILITY DAMPING DEVICES:		
ACOUSTIC CAVITIES	0.25" WIDE X 1.30 DEEP	NONE
BAFFLES	None	8 Blades
INJECTION ELEMENT TYPE:	"I" TRIPLET	"I" TRIPLET
NO INJ. ELEMENTS	72	168
THRUST/ELEMENT, (Lbf)	45 OUTER ROW 38 INNER ROW	45 UNIFORM

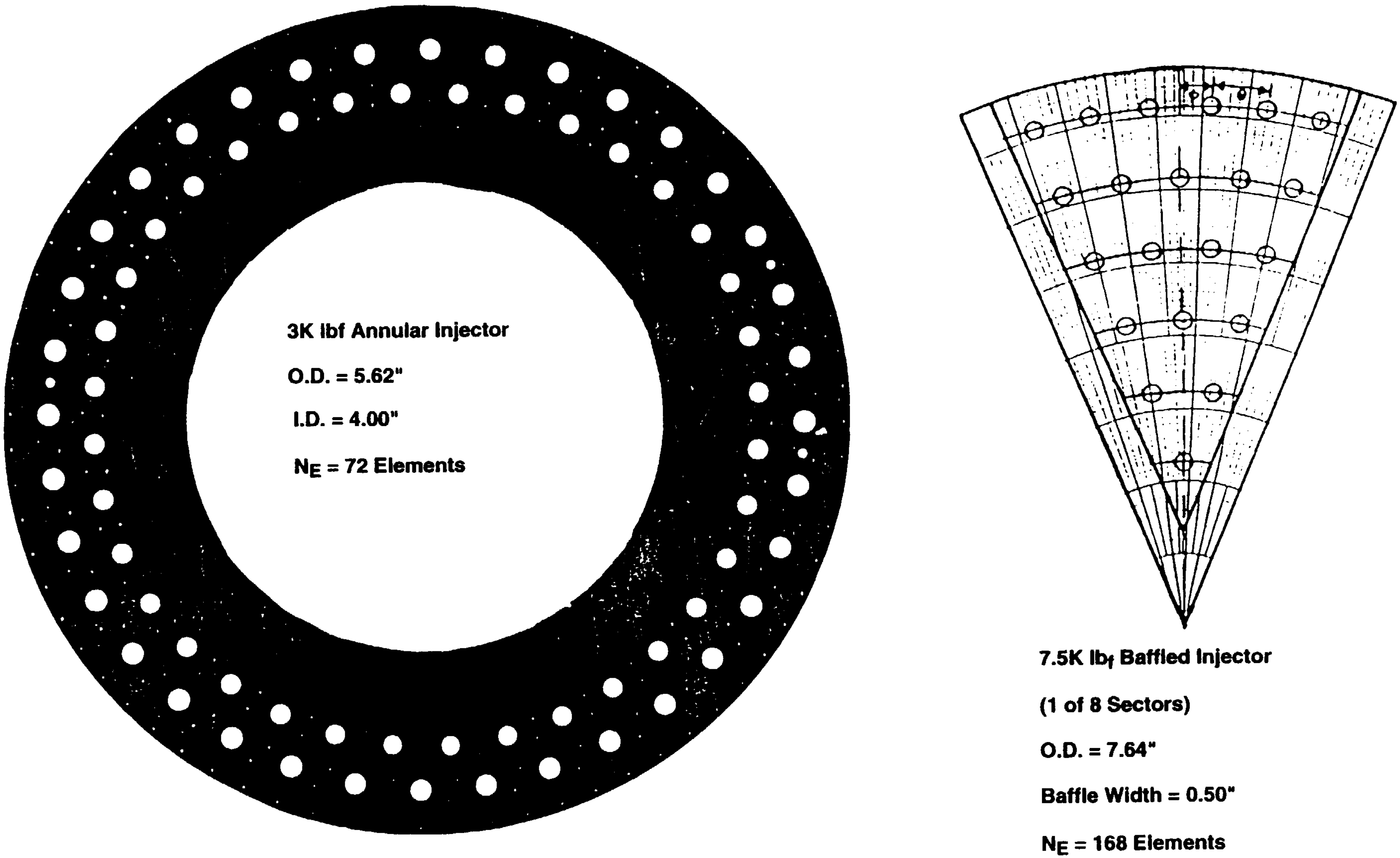


Figure 3.5.5-6 Uprated Injector Design Comparison

Injector design goals are listed in Table 3.5.5-2. These goals were selected to achieve the overall operational goals defined for the OTV engine. Use of gaseous phase propellants was selected as the best approach for providing the capability for wide range throttling and Mixture Ratio (MR) operations.

Specific design parameters for the injector are listed in Table 3.5.5-3. These parameters were identified by performing a system analysis of the OTV engine. The overall design goals are utilized to determine specific information on flowrates, pressure and temperature of the propellants being delivered to the injector. Nominal operational parameters for thrust, chamber pressure, specific impulse, and mixture ratio were selected as the design point. Dimensions of the injector face were determined by the OTV engine design, specifically: chamber design, chamber contraction ratio, and injector element.

The injector assembly is designed to accept gasified fuel from an external line routing from the fuel turbine exhaust. Exiting from this line, the fuel enters a common annular manifold as shown in Figure 3.5.5-7. Thru holes between the bolts, connect this annular distribution manifold to a secondary annular distribution manifold within the bolt circle. From this distribution manifold, fuel passages are electron discharge machined (EDM'd) between the oxidizer downcomers. Slots are depth cut into the manifold face. Thru drilled holes connect the passages to the slots. The redundancy in distribution manifolds is only to accommodate the test bed hardware. For the actual flight hardware, the bolts and outer distribution manifold would be eliminated thus reducing the overall external diameter and weight. Welding of the components would provide the leak-free joining mechanism.

An external line carries the gasified oxidizer from the oxidizer turbine exhaust into the oxidizer dome which encompasses the top of the injector. A splash plate is included to uniformly distribute the oxidizer prior to routing through the oxidizer posts. In this design, the oxidizer posts are thru drilled holes in the injector manifold. This eliminates internal brazing and interpropellant leakage paths.

The element pattern is contained in a stack of Nickel platelets which are diffusion bonded to the manifold. Platelets from the 3K lbf thrust level TCA which contain this "I"-triplet pattern are shown in Figure 3.5.5-8. The "I" triplet pattern was previously illustrated in

TABLE 3.5.5-2

INJECTOR DESIGN GOALS

• Energy Release Efficiency	>99.5%
• Throttling	10:1
• Mixture Ratio (MR)	5-7
• Uniform and Predictable Combustion Chamber Heat Flux at High Contraction Ratio (24.9:1)	10 Btu/sec in.2
• Stable Combustion	CPIA 247*
• Pressure Drop	<15% of Pc
• Cycle Life	500 Cycles
• Operating Life	20 Hours
• Fabrication	Low Cost
• Proof Pressure	3500 psi

*Compliance with Combustion Guidelines Chemical Propulsion Information Agency (CPIA)

TABLE 3.5.5-3

OTV INJECTOR DESIGN PARAMETERS

I. DESIGN PARAMETERS (Nominal)

A. Thrust, lbf (vacuum) - 7500

B. Chamber Pressure, psia - 2000

C. Propellants

1. Fuel - GH_2

a. $\dot{\omega}$ = 2.19 (lb/sec)

b. P = 2530 (psig)

c. T = 768 (°R)

2. Oxidizer - GO_2

a. $\dot{\omega}$ = 13.79 (lb/sec)

b. P = 2645 (psig)

c. T = 784 (°R)

D. M.R. = 6.0

E. Injector Face O.D. = 7.64"

F. Isp = 480 sec

G. Baffles

1. 8 Blades

2. 0.5" thick x 5" axial length (from injector face)

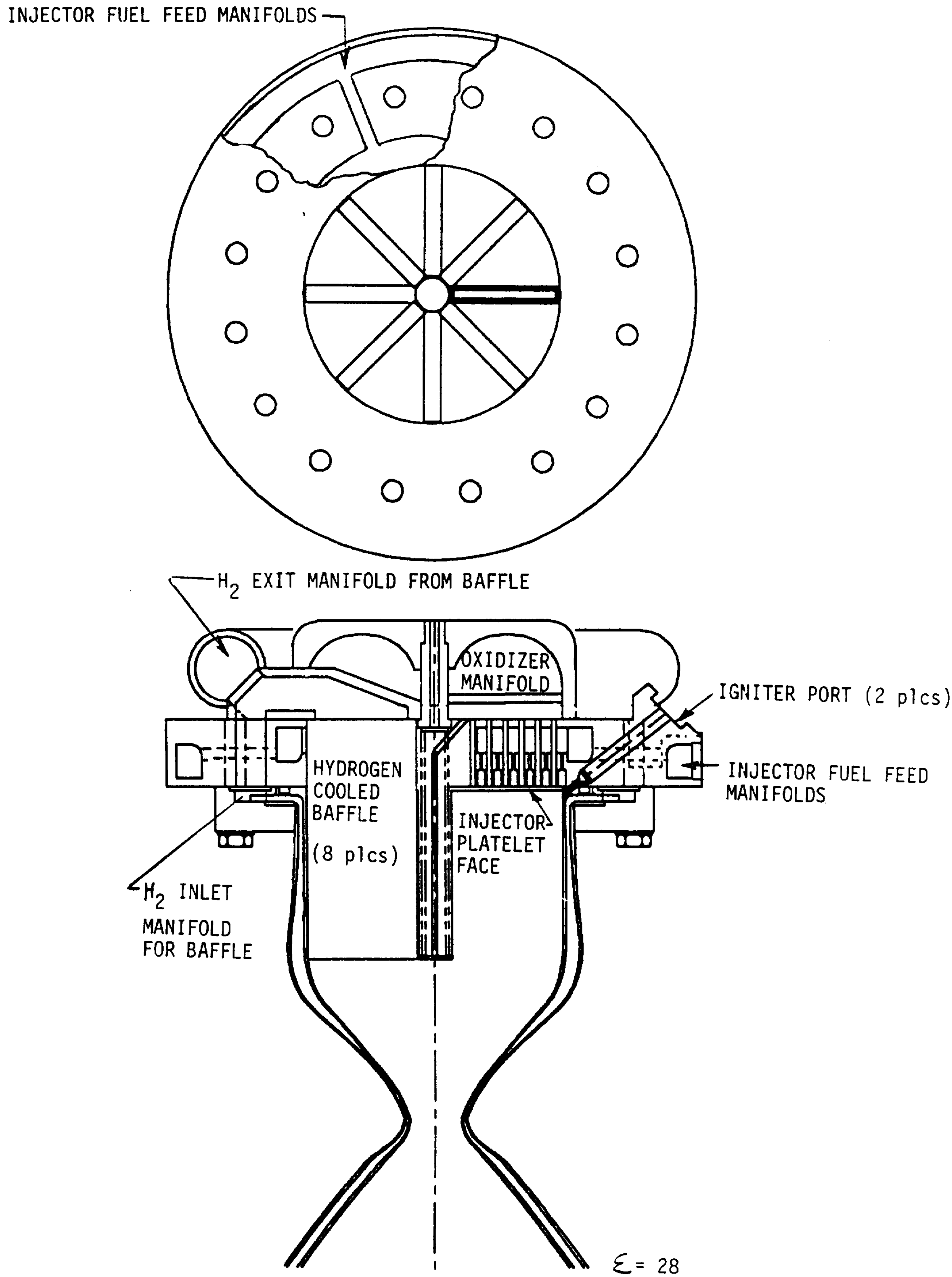


Figure 3.5.5-7 7.5K lbf OTV TCA

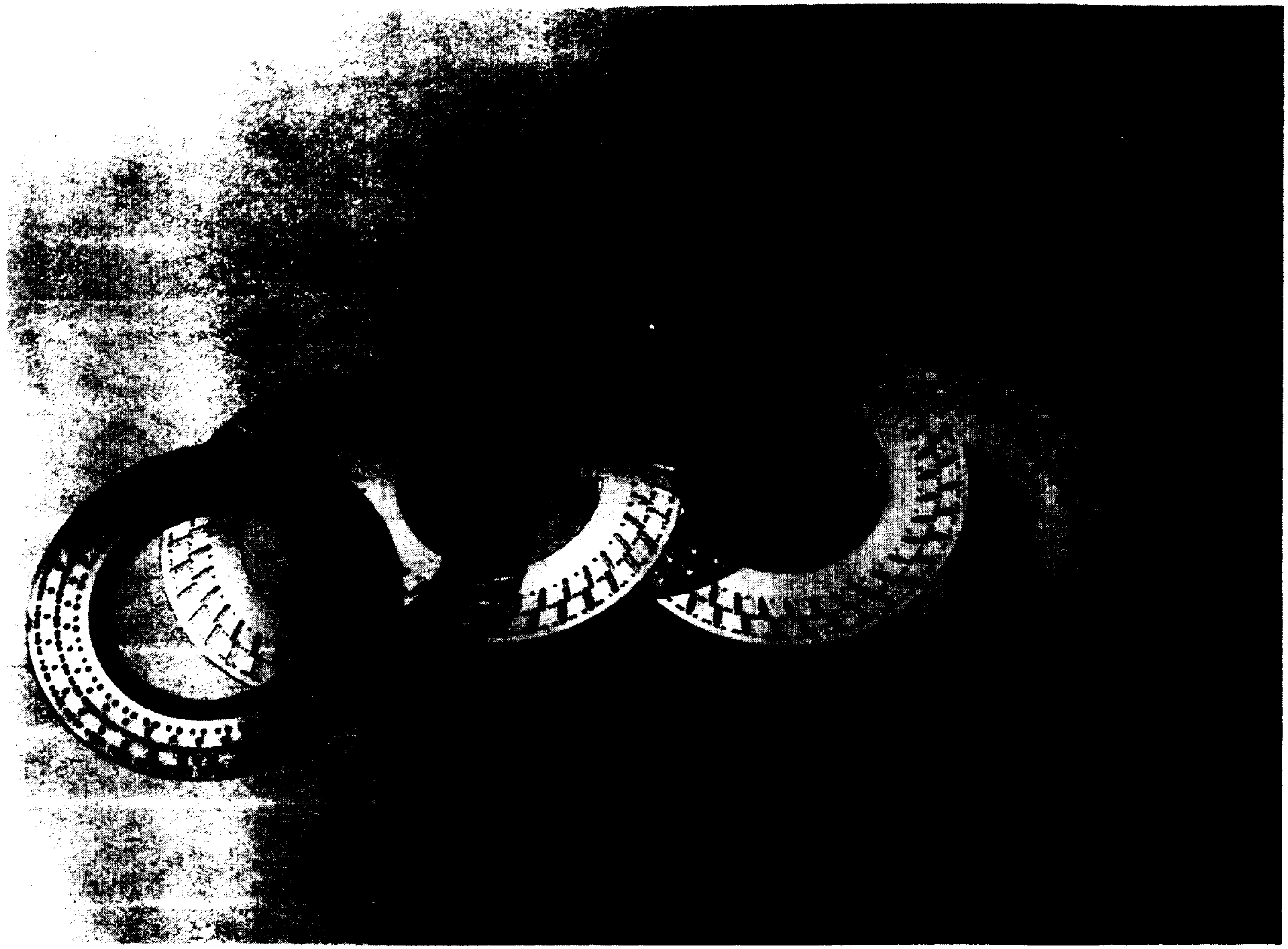


Figure 3.5.5-8 Injector Platelets

Figure 3.5.5-3. The fuel passages are routed and put through a 90° bend to impinge on the oxidizer stream. A cup provides an area for mixing of the fuel and oxygen prior to injection into the chamber for combustion.

Finalization of the element design is dependent on the results of the hot fire testing of the 3K lbf thrust level injector. Because the dual propellant expander cycle depends on a specific heat input into the regen cooled components, verification of this value is necessary. Fine tuning can be done to the element dimensions in the design phase if the nominal heat flux levels are not met.

3.5.5.1.3 Regenerative Cooled Baffles

The 8-baffle plates are designed as a separate integral assemblies as shown in Figure 3.5.5-9. Hydrogen passages are chemically etched in ZrCu platelets which are diffusion bonded together (Figure 3.5.5-10). The baffle plates are cooled with hydrogen which flows down one side, across the radiused bottom, and back up the opposite side. The two opposing ends of the baffles are cooled in a similar manner. Common manifolding in the baffle plate feeds the inlets to the side and end passages (Figure 3.5.5-11). Exiting hydrogen is collected in a separate common manifold adjacent to the inlet. External lines are connected to these manifolds for further routing of the fuel.

The baffles are fabricated from ZrCu platelets which are diffusion bonded or diffusion brazed together. Each blade is 0.5 inches thick, 3.13 inches wide (radially) to the point of contact between two adjacent blades, and 5.19 inches long. The last 0.25 inches of baffle axial length are radiused in order to minimize fluid flow pressure drop in the coolant turnaround region.

Two methods of support and inlet/outlet coolant flow paths are possible: from the injector face or from the chamber walls. The structural complexity and uncertainty associated with the attachment of the baffle assemblies to, and coolant flow through, the chamber walls directed the design toward an extension from the injector face.

Attachment to the injector and inlet/outlet manifolding was approached by having the baffle assembly a completely sealed unit to avoid leakage between the coolant and the propellants.

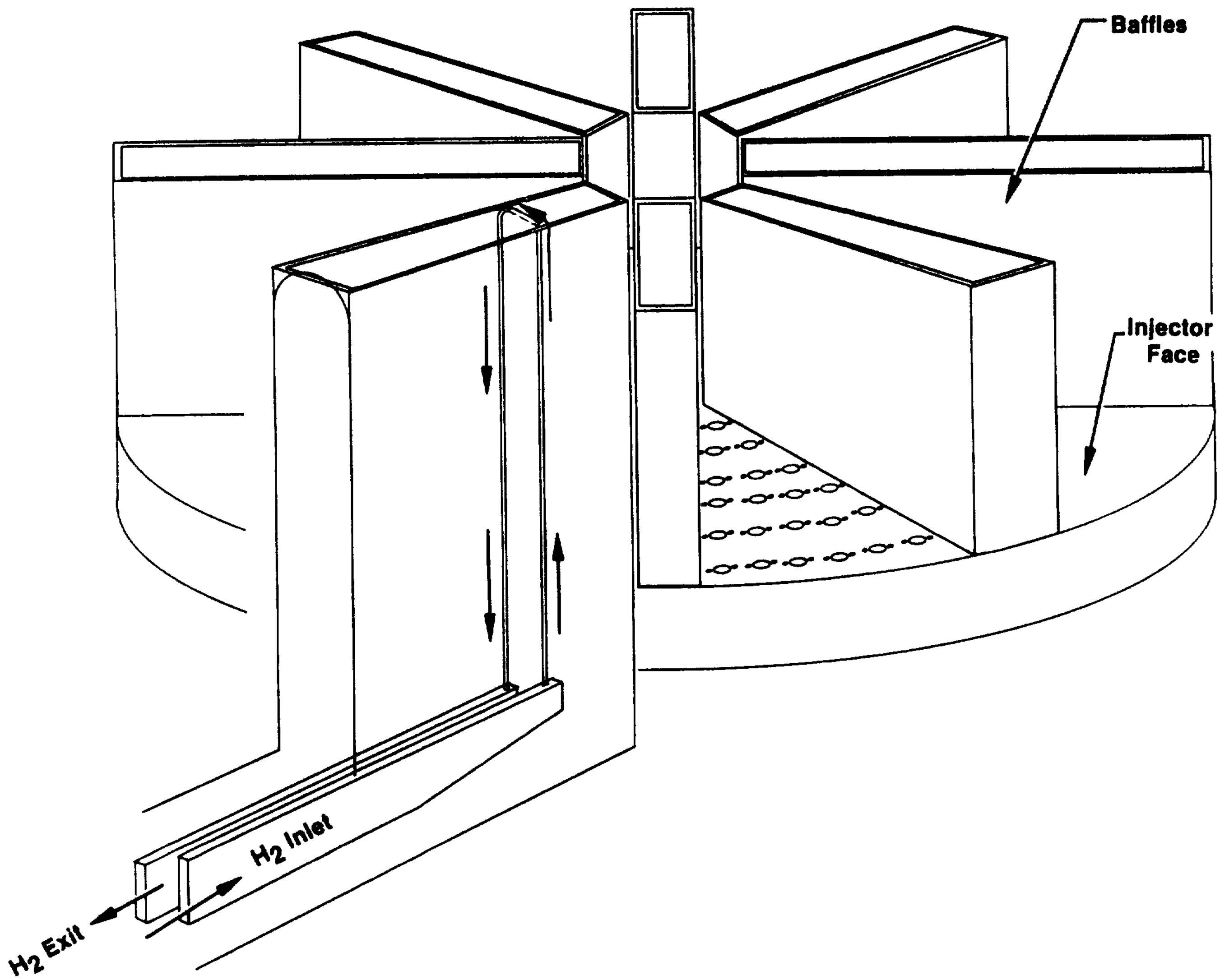


Figure 3.5.5-9 Baffle Assembly

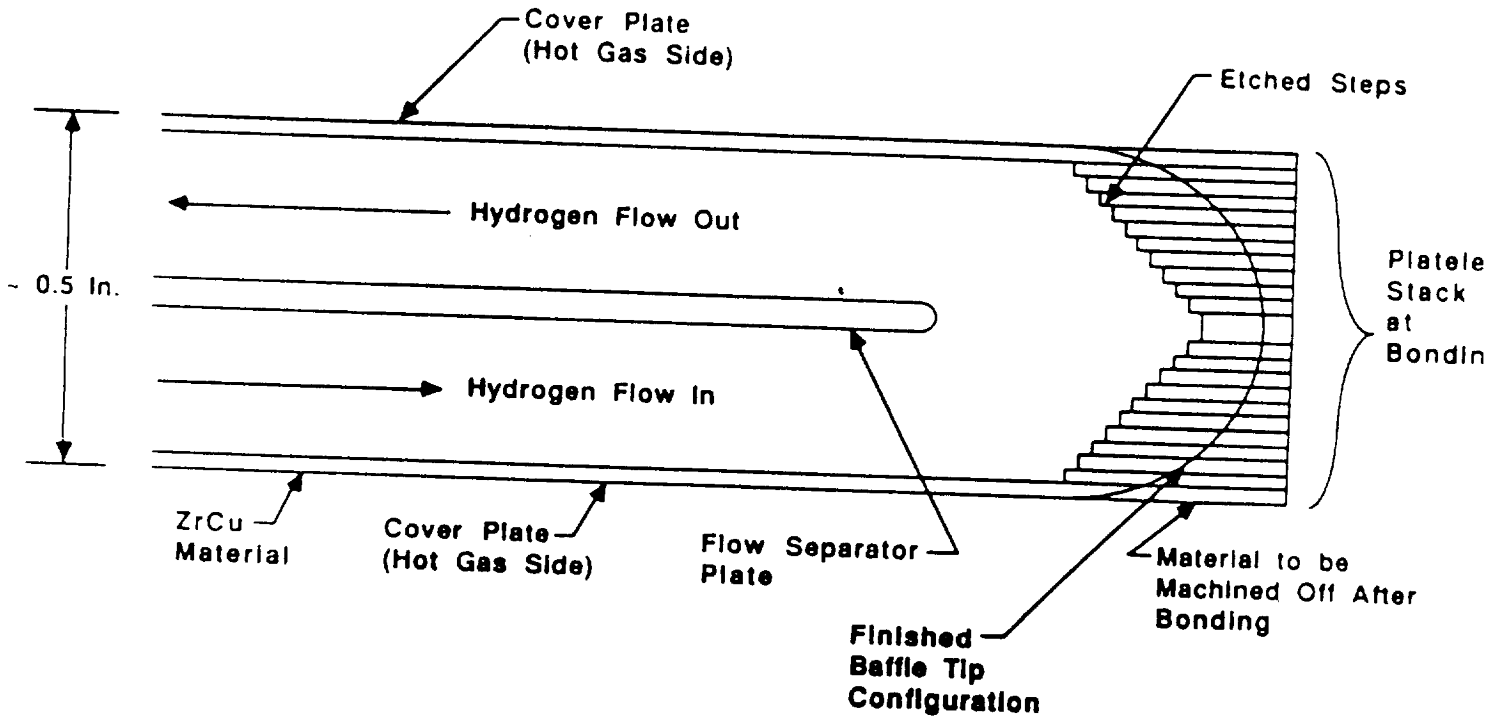


Figure 3.5.5-10 Diffusion Bonded ZrCu Platelets

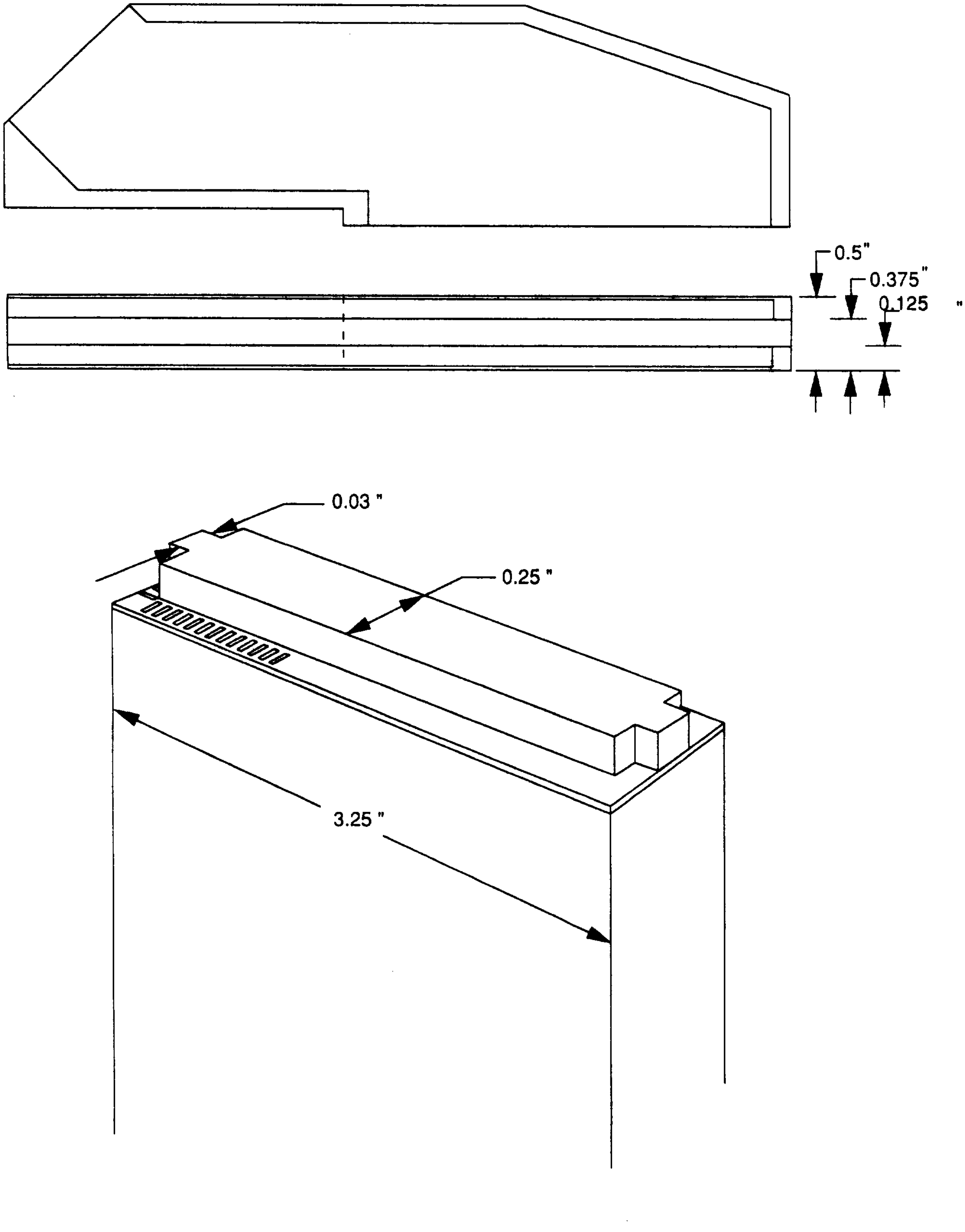


Figure 3.5.5-11 Baffle Manifolding Details

The injector manifold has mating slots cut for the plates to be slid into place from the backside. Brazing of the contacting surfaces of the baffle and the manifold slots seal the units in place. Mechanical seals prevent leakage from the flange inlet port to the baffle and a welded junction prevents leakage as the baffle exits into a common manifold.

3.5.5.1.4 Chamber

The flow circuitry consists of a single pass of hydrogen flowing counter to the combustion gas stream. The hydrogen enters at an area ratio of 28:1 and exits at the injector face plane. A total of 233 cooling channels is proposed for the regen cooled jacket. The channel inlet (at an area ratio of 28:1) must accommodate additional heating due to the presence of the flange. However, no heating problem is anticipated due to the low hydrogen temperature and low heat flux in this region.

The coolant channels in the converging and diverging sections of the regeneratively cooled jacket are fabricated using a straddle mill cutter. This technique provides a smooth transition in channel width from the value at the throat to the values in the barrel and nozzle. The L' (injector face to throat distance) is 9.8 inches. The radius of curvature normalized to the throat radius for the start of the converging section; upstream of the throat; and downstream of the throat is 2.0, 1.5, and 1.5, respectively. The convergence angle is 40 degrees. The gas side wall material is NASA-Z copper. The back side material is NiCo. The chamber barrel diameter is 7.64 inches. the throat diameter is 1.526 inches.

The channels adjacent to the baffle edges in the barrel section will not see the same heat load as the channels adjacent to the hot gas flow field therefore the outlet hydrogen temperature of 820°F represents a mixed mean hydrogen temperature. Because the hydraulic resistance for the heated and unheated channels in the barrel section of the regen cooled jacket is small compared to the rest of the chamber, the flow maldistribution is considered negligible.

3.6 SPACE BASING CONSIDERATIONS

The OTV can be ground based or space based. On a ground based vehicle maintenance can be accomplished after every mission, often in a shirt-sleeve environment. A full complement of maintenance equipment can be readily available in a specially designed facility. Under such conditions the engine design can be biased towards integration with the vehicle and low weight at some sacrifice in ease of maintenance access. With a space based system there is a premium on reliability and a design emphasis on access to selected components by personnel in spacesuits. Aerojet has assumed that the Orbit Transfer Vehicle will be space based with routine servicing performed in a maintenance hangar at the Space Station. An alternate location in the 21st century is maintenance at a Lunar base. The maintainability concept outlined in Section 3.6.3 assumes space basing.

Space basing and ease of vehicle/engine checkout are also criteria for the health management approach discussed in Section 3.6.2. The effects on the engine design from the space environment itself are discussed in Section 3.6.1.

A summary of space basing considerations is given in Table 3.6-1. Only those most important to the engine design are discussed in the sections that follow.

3.6.1 Space Environment Effects

3.6.1.1 Vacuum Operation

The most immediate exploitation of space operation is the design of the engine without a purge system. Mechanically separate turbopumps using only hydrogen or oxygen avoid the need for a turbopump purge. The ready dissipation of the cryogenic propellants in space prevents concentration in densities that would lead to flash fires or explosion during TPA chilldown and engine start and shutdown. This is a major reduction in engine system weight and complexity.

A secondary benefit of vacuum operation is the ability to design a leak detector that will pick up very low pressures of oxygen or hydrogen in the engine compartment and warn of a leak before it progresses to a dangerous state. This is a difficult task in an air environment as most detectors are unable to discriminate between gases at normal concentration in the air and those progressing to an abnormal concentration.

Table 3.6-1

OTV Engine Preliminary Design Space Basing Considerations

	SPECIFIC CONCERN
• IMPACT OF ENVIRONMENTAL EFFECTS ON ENGINE DESIGN	
— RADIATION	
MATERIALS	POLYMERS, PRECIPITATION HARD CU ALLOY
ELECTRONICS (SINGLE EVENT UPSETS)	SENSORS, CONTROLLER
— MONATOMIC OXYGEN	NONE
— METEOROID	EXTENDIBLE NOZZLE
— TEMPERATURE	COMPONENT THERMAL CONDITIONING
— VACUUM	APPROVED MATERIALS LIST
• HEALTH MANAGEMENT APPROACH	
— WEAR PATTERNS AND FAILURE SIGNATURES	VALVES AND TURBOPUMPS
— HEALTH MONITOR SENSORS	RELIABILITY, SPACE CHANGEOUT
— IMPROVED SENSORS (CONVENTIONAL MEASUREMENTS)	REDUCED SIZE, WEIGHT
— DIAGNOSTICS	EXPERT SYSTEM SOFTWARE
— PROGNOSITICS	EXPERT SYSTEM SOFTWARE
• SPACE-BASED ENGINE INSPECTION AND CHECKOUT	
— REQUIREMENTS	MINIMIZE EVA, DEVELOP SOFTWARE
— TOOLS	SPECIALIZED KIT, AUTOMATED
— TECHNIQUES	AUTOMATION, COMPUTER EXPERT SYSTEM
— TIME LINES	RAPID TURNAROUND

3.6.1.2 Materials Selection

High performance LO_2/GH_2 engines generally use a copper alloy for the combustion chamber. Copper alloys are also candidates for cooled nozzle extensions and heat exchangers. One of the concerns in materials selection for space use is the suitability of copper alloys for structures exposed to the radiation environment. Radiation exposure can change the structural bonds and physical properties of materials. Table 3.6.1-2 summarizes the effects of two levels of radiation exposure on various copper alloys. In the column under yield strength (YS) note that oxygen free (OF) copper actually improves in strength after a 15 dpa exposure,* DS copper C15720 (an older GLIDCOP formulation) is relatively unchanged, and zirconium copper (Zr-Cu), has a dramatic reduction in yield strength. Zr-Cu is typical of precipitation hardened copper alloys, and NASA-Z copper should show a similar change in properties. Also, the last column shows the area reduction of samples under radiation exposure. Note that OF copper test sample shrink dramatically as does those of Zr-Cu. Structural shape, welds, and mechanical fasteners are all suspect under such a dramatic change in physical size. The opposite effect, a swelling of the copper alloy, is documented in Table 3.6.1-3 for neutron irradiation. Again, OF copper and precipitation hardened coppers such as NASA-Z and Zr-Cu are most affected. A dispersion hardened copper such as DS C15720 is affected but less severely.

The total radiation exposure is dependent on the type of mission and the number of years operation in space. Transit through the VanAllen belts and Lunar missions will greatly increase exposure compared to low earth orbit operation. Exposure to one large solar flare could give a radiation dose equivalent to years of normal operation. The prudent design course is to use copper alloys relatively impervious to radiation. The DS coppers (GLIDCOPs) and Cu-Mg-Zr-Cr copper are the best candidates identified to date.

3.6.2 <u>Health Management Approach</u>

3.6.2.1 Health Monitor and Control Sensors

The engine health monitor and control sensors are shown on Figure 3.6.2-1 and listed in Table 3.6.2-1. Also, Table A-2.2 in Appendix A-2 lists the following information for each sensor:

- Sensor identification
- Component on which the sensor is installed
- Location of sensor on component

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*SCM Metal Products Technical Bulletin #1428

<u>TABLE 3.6.1-2</u>

OTV ENGINE MATERIALS SELECTION

- Thrust Chamber Assembly in a High Radiation Environment

Tensile Properties of Neutron Irradiated Copper Alloys

Material/Condition		YS (MPa)	UTS (MPa)	Elongation (%)	Reduction in Area (%)
Copper:					
Marz:*	Control	31	152	24	89
	3 dpa	33	155	25	80
	15 dpa	41	149	17	48
OF:	Control	26	196	27	90
	3 dpa	51	200	30	59
	15 dpa	49	185	24	46
DS Copper:					
C15720:	Control	337	395	14	48
	3 dpa	353	411	14	51
	15 dpa	343	395	20	50
C15760:	Control	397	466	12	56
	3 dpa	402	468	10	43
	15 dpa	379	449	11	50
Precipitation-Hardened:					
Cu-Zr:	Control	271	334	9	70
	3 dpa	274	317	9	73
	15 dpa	71	226	34	44
Cu-Mg-Zr-Cr:	Control	431	458	10	61
	3 dpa	379	437	13	56
	15 dpa	284	354	17	57

*Marz - Commercial name SCM Metal Products, Cleveland, Ohio.

TABLE 3.6.1-3

OTV ENGINE MATERIALS SELECTION

- Thrust Chamber Assembly in a High Radiation Environment

Swelling of Neutron Irradiated Copper Alloys

Material	Volume % Increase After Irradiation	
	3 dpa[1]	15 dpa[2]
Copper		
Marz grade (99.999%)	1.8	6.8
OF grade (99.95%)	2.1	6.6
DS Copper:		
C15720	0.8	0.9
C15760	1.1	0.6
Precipitation Hardened:		
Cu-Zr	nil	3.6
Cu-Mg-Zr-Cr	nil	nil

[1] 3 dpa corresponds to fluence of 0.4×10^{26} n/m^2 (F.n > 0.1 MeV)
[2] 15 dpa corresponds to fluence of 2.0×10^{26} n/m^2 (F.n > 0.1 MeV)

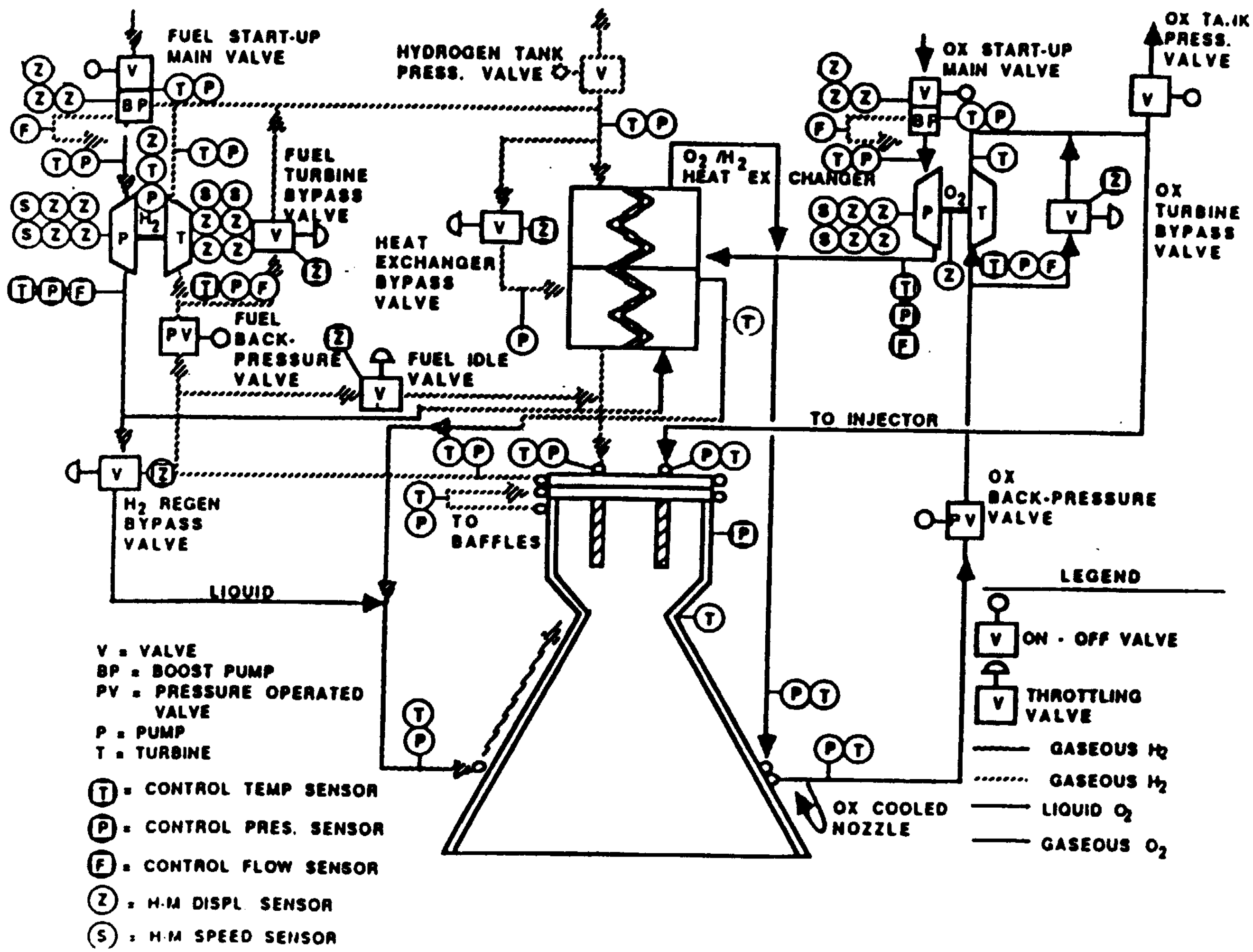


Figure 3.6.2-1 OTV Engine Preliminary Design Dual Expander Cycle Sensor Locations

TABLE 3.6.2-1

OTV ENGINE SENSORS

Parameters	Function
1. Turbopump Sensors	
• Shaft Displacement (3 Axes)	HM-1
• Speed	HM-1
• Pump Discharge Pressure	Control
• Pump Discharge Temperature	Control
• Pump Flow	Control
• TPA Vibration	HM-1
• Pump Interstage Pressure	HM-2
• Pump Interstage Temperature	HM-2
• Turbine Interstage Pressure	HM-2
• Turbine Interstage Temperature	HM-2
• Turbine Inlet Temperature	Control
• Turbine Inlet Pressure	HM-1
• Turbine Inlet Flow	HM-1
• Pump Inlet Temperature	HM-2
• Turbine Discharge Pressure	HM-2
• Turbine Discharge Temperature	HM-2
2. Boost Pump Sensor	
• Bearing Outer Race Deflection/Speed	HM-1
• Vibration	HM-2
• Pump Inlet Pressure	HM-2
• Pump Inlet Temperature	Control
• Turbine Inlet Flow	HM-2
3. Combustion Chamber Sensors	
• Chamber Pressure	Control
• Coolant Outlet Pressure	HM-1
• Baffle Coolant Outlet Pressure	HM-2
• Baffle Coolant Outlet Temperature	HM-2
• Throat Surface Temperature	HM-2

TABLE 3.6.2-1

OTV ENGINE SENSORS (Cont.)

Parameters	Function
4. Nozzle Sensors	
• Fuel Coolant Inlet Pressure	HM-1
• Fuel Coolant Inlet Temperature	HM-1
• Oxidizer Coolant Inlet Pressure	HM-1
• Oxidizer Coolant Inlet Temperature	HM-1
• Oxidizer Coolant Outlet Pressure	HM-2
• Oxidizer Coolant Outlet Temperature	HM-2
5. Regenerator and Injector Sensors	
• Fuel Regenerator Outlet Temperature	Control
• Fuel Regenerator Inlet Temperature	HM-2
• Fuel Regenerator Inlet Pressure	HM-2
• Injector Fuel Inlet Pressure	HM-1
• Injector Fuel Inlet Temperature	HM-1
• Injector Oxidizer Inlet Pressure	HM-1
• Injector Oxidizer Inlet Temperature	HM-1

- The sensor function
- Sensor weight
- Power requirements

The sensor selection is based on the technology that is expected to be available at the time of the engine system development.

The sensor functions are divided into three categories in order of priority; Control, Primary Health Monitor (HM-1), and Second Level Health Monitor (HM-2).

Control sensors monitor critical parameters essential for engine control. Redundant sensors are recommended for each of the eight control parameters. These sensors also provide information to the health monitor system where applicable.

The primary health monitor sensors (HM-1) monitor critical component conditions and provide system status such as injector inlet conditions and chamber/nozzle cooling.

The second level health monitor sensors (HM-2) monitor less critical component parameters which are used to detect component degradation and to provide system and component performance assessment.

3.6.2.2 Sensor Configuration and Type

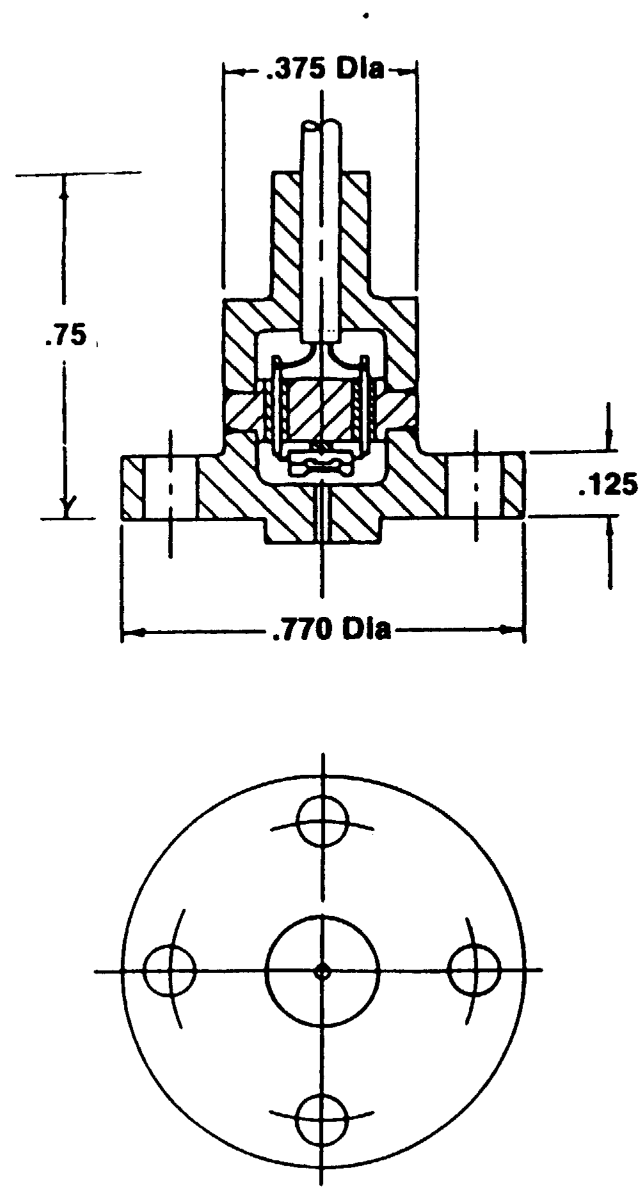
The basic sensor transducer types selected for the OTV engine are listed in Table 3.6.2-2. The transducer types are either current state-of-the-art or under development. NASA LeRC is funding the development of the three function capacitive displacement sensor under Contract NAS 3-23772. The leak detector, specifically to detect low pressure hydrogen and oxygen, is under development as a separate task on the same contract. The valve position sensor is an application of the displacement sensor design. All other sensors are available in a form usable for the OTV engine in the near future.

Specific sensor types are shown in Figures 3.6.2-2 through 3.6.2-8. These were used in the component design to assure that sensors were baselined in the design and not a retrofit.

TABLE 3.6.2-2

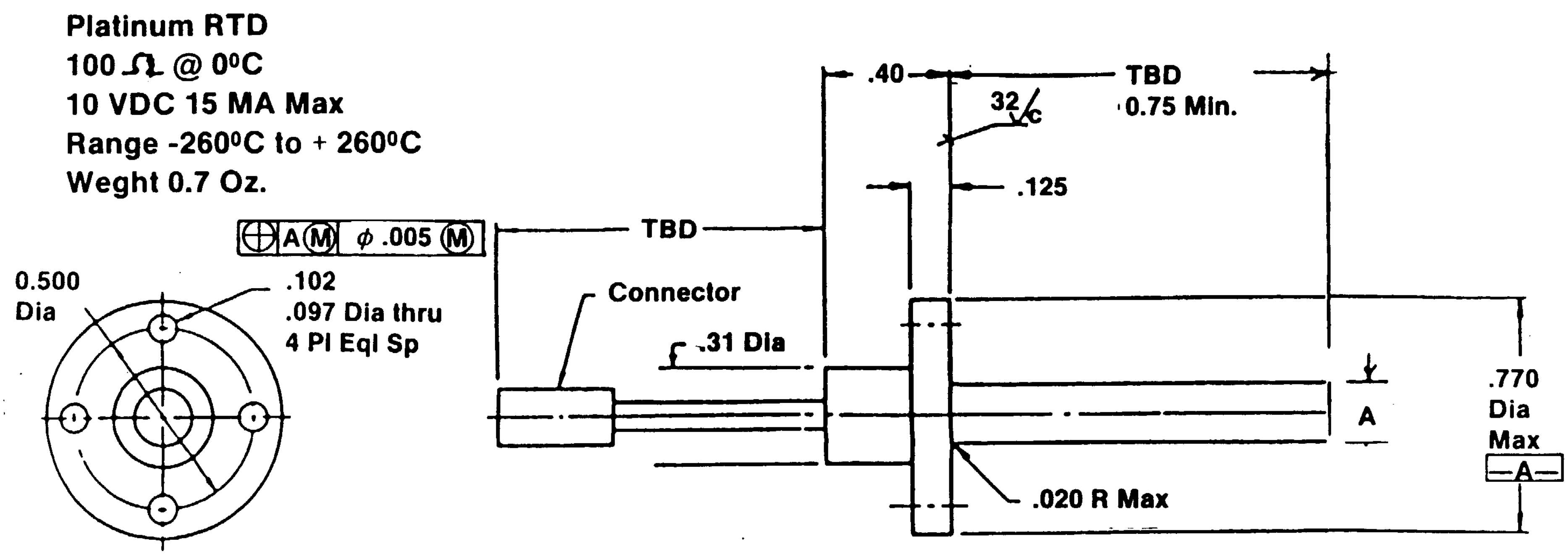
SENSOR TYPES

- Pressure - Silicon capacitive sensors for cryogenic temperature operation.
- Temperature - 100 ohm Platinum Resistance Temperature Detector (RTD)
- Flow - Vortex shedding with Piezoelectric sensing
- Shaft Displacement/Speed - Three function capacitive displacement sensor
- Bearing Race Deflection - Capacitive Displacement sensor
- Valve position - Capacitive Displacement sensor (redundant sensors)
  - Microswitch (for Open/Close Valves)
- Acceleration - Piezoelectric sensor with built in preamplifier
- Leak Detection - Distributed palladium based chemical gas detectors



Excitation	10 Volts DC 15 MA
Iniput. Res.	1000Ω Min
Output	30 MV DC
Weight	0.5 Oz
Oper. Temperatrue Range	-320ºF to +150ºF

Figure 3.6.2-2 Pressure Sensor



Single Element "A" = 0.125
Dual Element (Redundant) "A" = 0.200

Figure 3.6.2-3 Temperature Sensor - Immersion Type

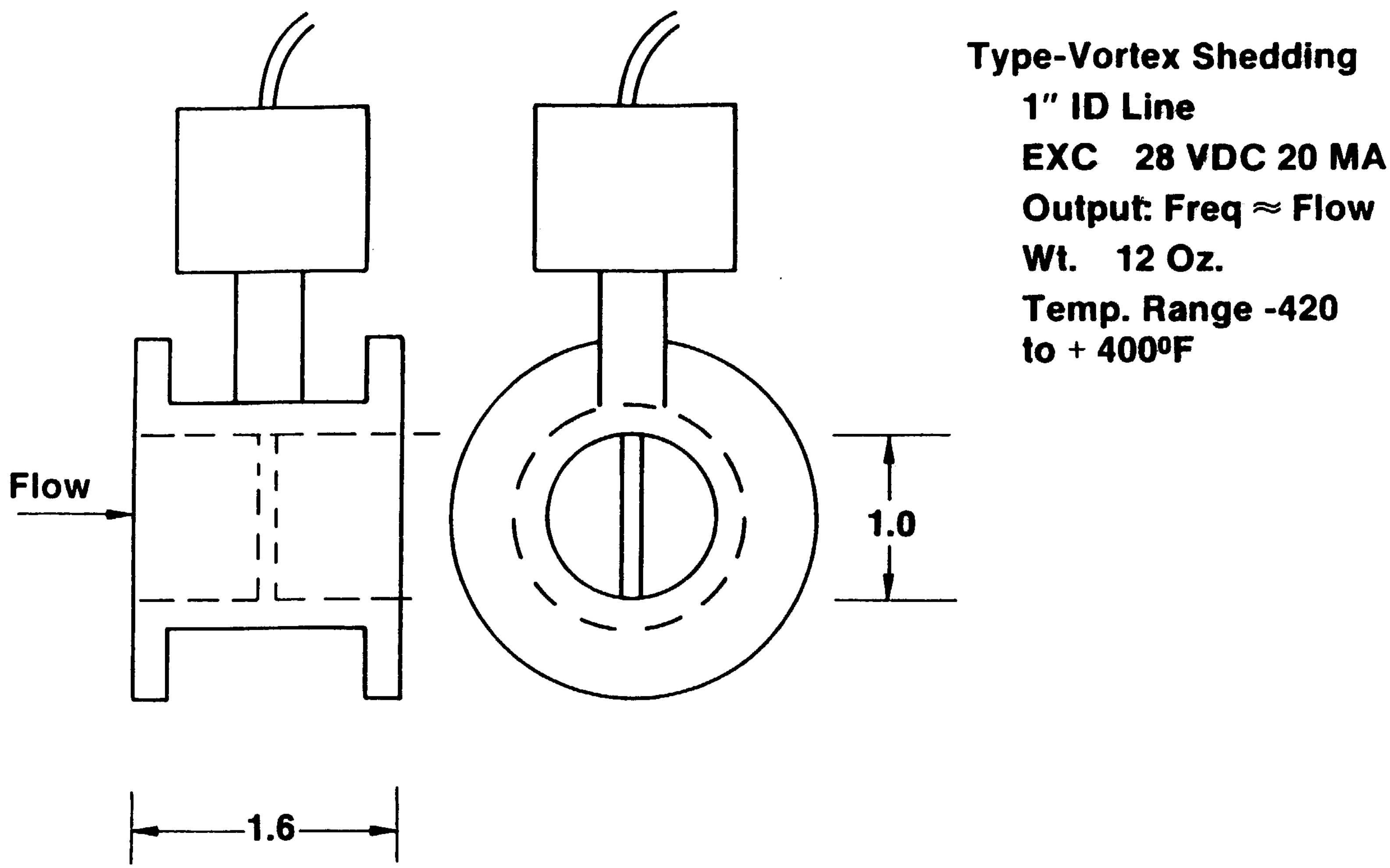
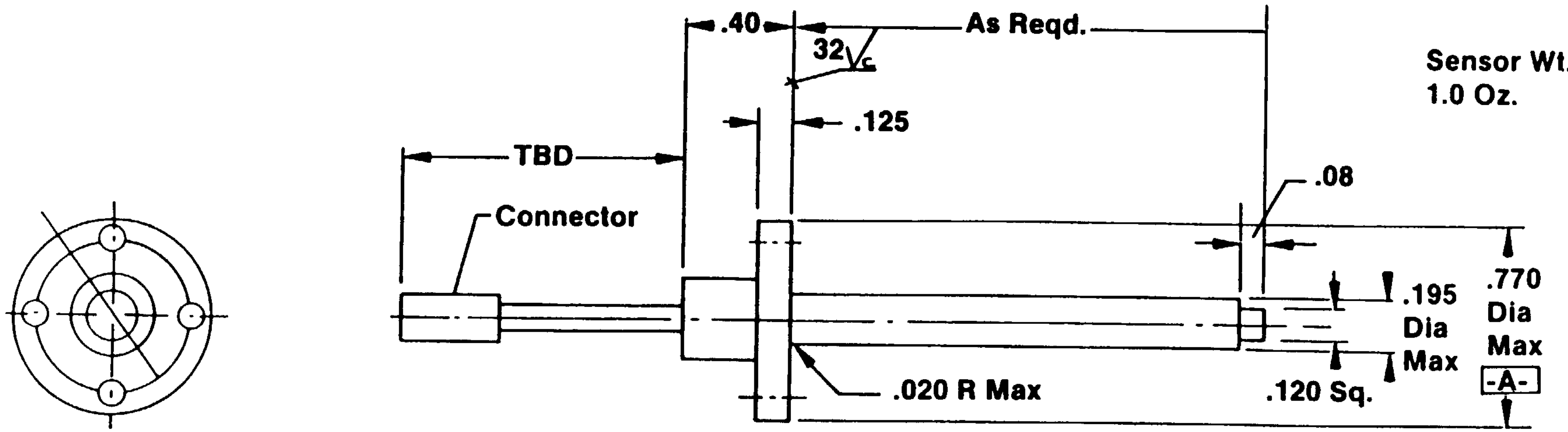


Figure 3.6.2-4 Flow Sensor

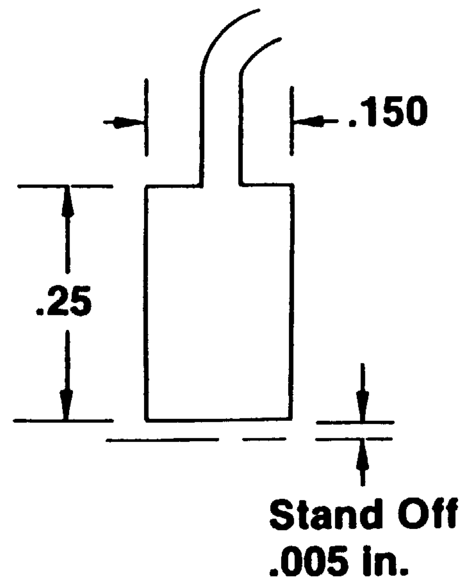


Requires Oscillator/Driver 8.0 In. Max from Sensor
Size 0.5 x .75 x 1.5
Wt. 1.5 Oz.
Input Power 10 Volts DC 30 MA

Measures
Axial Displacement
Radial Displacement
Speed

Figure 3.6.2-5 Displacement/Speed Sensor

Capacitive Sensing



Wt. 0.02 Oz.

Requires

Oscillator/Driver 8" Max from Sensors
Size 0.5 x .75 x 1.5 in.
Wt. 1.5 Oz.
Serves 2 Sensors
Power 10 Volts DC 20 MA

Figure 3.6.2-6 Bearing Displacement Sensor

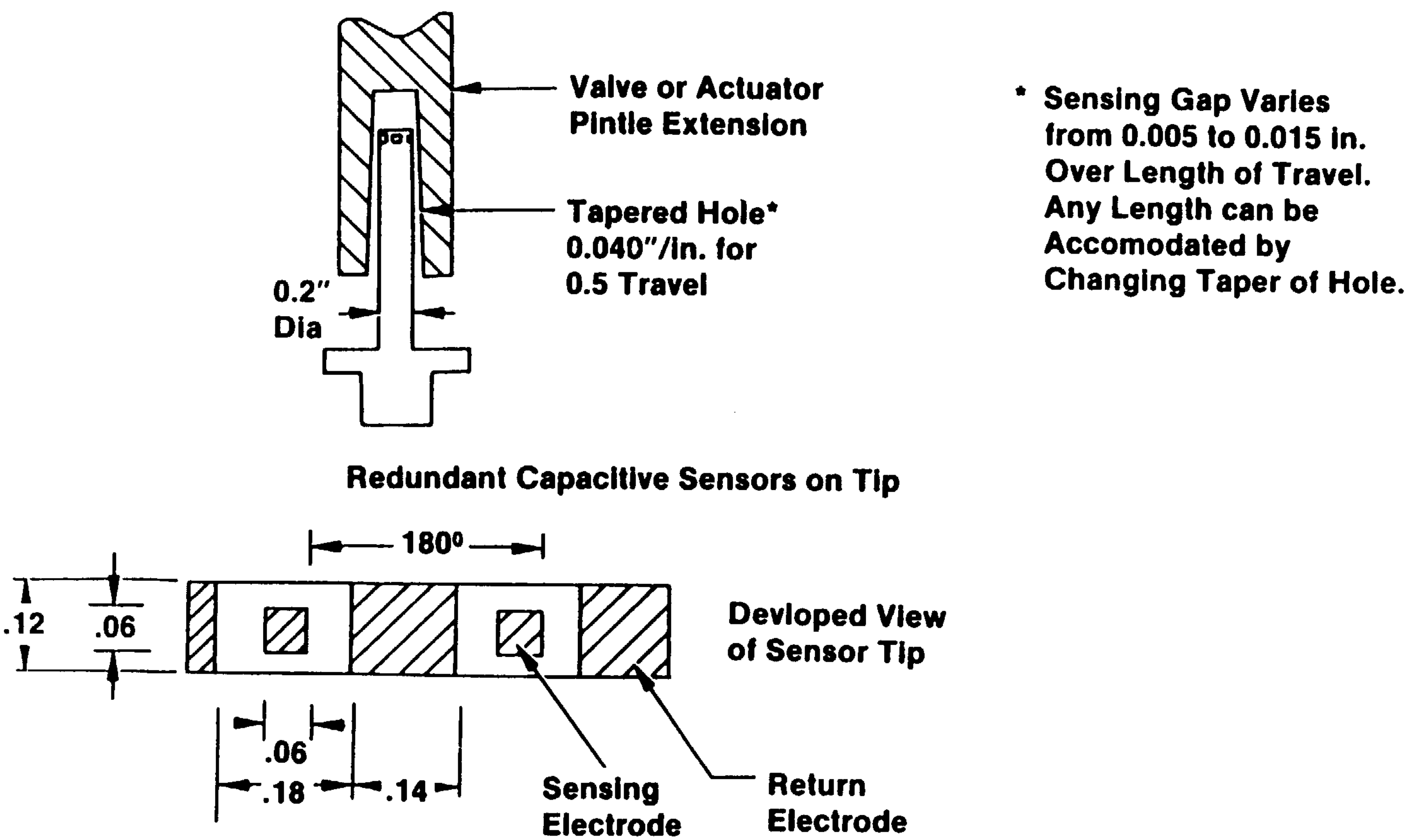


Figure 3.6.2-7 Capacitive Linear Displacement Sensor

SPECIFICATIONS

PERFORMANCE

Sensitivity	10 mv/g nominal
Frequency range ±5%	3Hz - 10kHz
Mounted resonance	90 kHz nominal
Gmax (w/o clipping)	250
Noise Floor (nominal)	.002 g (rms)
Max shock w/o damage	10,000 g
Transverse sensitivity	5% max.
Temperature Sens (typical)	
−65°F to +100°F/36°C	+5%
+100°F to 150°F/66°C	−5
+100°F to +250°F/120°C	−8
Output Impedance (ohms)	1000
Linearity - 200 g	±2%

PHYSICAL

Weight (grams)	2000	2.1 = .07 oz
	2001	1.9
Wiring		2 wire coaxial
Power		9-24 Volts DC
		0.5-4ma
Grounding		2000 - Isolated
		2001 - Case Ground
Connector		Coaxial
Mounting stud		6-32, 1/8
Material		Titanium

OUTLINE DIMENSIONS

2000/2000A
Isolated

2000A, 2001A
without mounting
stud (for adhesive
mounting)

2001/2001A
Case Ground

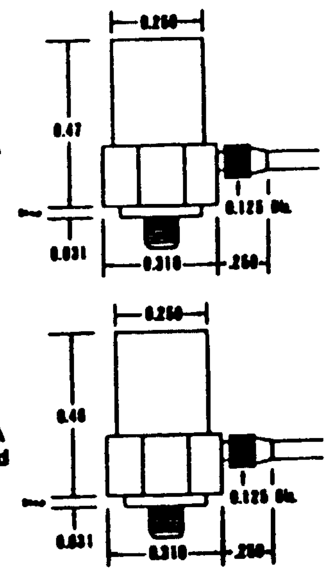


Figure 3.6.2-8 Accelerometer

3.6.2.3 Health management

Task E.3 under contract NAS 3-23772 developed an integrated approach to the engine control and health monitoring tasks. A comprehensive final report on the task was issued in October 1988 as NASA CR-182122. It is listed in the references as Reference 6. It has a detailed discussion on health management in the space-based environment. The basic requirements of a health monitoring system are:

- Institute corrective action to prevent engine failure
- Correct off nominal operation
- Safely shutdown the engine when there is a component failure
- Provide engine status on pre- and post-flight
- Record operating data for later preventive maintenance action
- Alert vehicle flight crew of potential engine or component problems
- Provide data readouts to assist in troubleshooting
- Provide information for closed loop engine control.

The benefits expected from such a system are:

- Enhance system reliability
- Extend system life
- Reduce life cycle costs
- Improve mission safety

The results of the Task E.3 work are summarized in Table 3.6.2-3. The additional work done in this preliminary design task focused attention on the actual sensors needed, identified potential sensors, and assured that component designers included provisions for the necessary sensors in the design. Also, weights and power requirements were estimated for use in determining engine weight and operating power consumption. The combination of work done under these two tasks has established a baseline for the engine detail design work in the controls and health monitoring area.

3.6.3 Maintainability

In-space maintenance is a complex subject. Some of the considerations are listed in Table 3.6.3-1. They cannot be addressed in detail without a better definition of the basing and overall vehicle design. The intent of the Aerojet maintainability analysis was to concentrate on

Table 3.6.2-3

OTV Engine Integrated Control and Health Monitoring Task

Summary of Results

- Nonlinear Dynamic Model Verified Throttling Capability Using Selected Controls and Sensor Input
- Proposed Control Logic Valid for Most of the Operating Range but Added Logic Needed for Low Thrust Operation
- An ATC Developed Controller was Evaluated and is Considered Readily Adapted to the OTV Engine
- Health Monitoring Requirements were Identified and a Specific Approach Recommended
- Health Monitor Sensors were Identified, Preliminary Specifications Developed, and a Supplier Survey Conducted

Results of This Task Were Used in Developing Ongoing Sensor Development Tasks

Table 3.6.3-1

OTV Engine In-Space Maintenance

CONSIDERATIONS

- COST EFFECTIVE DESIGN FOR A REUSABLE SYSTEM MUST INCLUDE MAINTAINABILITY
- MAINTENANCE COSTS INCLUDE
  - TRAINING
  - FACILITY
  - DOWNTIME
  - SPARES & SPARES STORAGE
  - TOOLS/DIAGNOSTIC DEVICES
  - PERSONNEL COSTS
  - INSPECTION/CERTIFICATION
  - ADMINISTRATION
- IN-SPACE MAINTENANCE POSES SEVERE MAINTAINABILITY PROBLEMS
  - SPECIALIZED TOOL KIT
  - LACK OF GRAVITY ASSISTANCE
  - LIMITED EFFECTIVE WORKING TIME
  - ACCESS PROBLEMS FOR MECHANICS IN SPACE SUITS
- RELIABILITY IN RECONNECTING FLOW LINES AFTER MAINTENANCE IS A FUNCTION OF LINE SIZE AND OPERATING PRESSURE
  - BEST BREAK POINT IS IN LARGE, LOW PRESSURE LINES
  - SMALL HIGH PRESSURE LINES HAVE HIGH LEAK POTENTIAL
- CONNECTIONS FOR HEALTH MANAGEMENT SENSORS COMPLICATE COMPONENT CHANGE OUT
- ACTUAL, DEMONSTRATED IN-SPACE REPAIR OPERATIONS HAVE BEEN FAR MORE TIME CONSUMING AND FATIGUING THAN EXPECTED

the engine and the components that could be changed out in space by astronauts in space suits. The components are listed in Table 3.6.3-2. The comments column contains frequent reference to access near the vehicle/engine interface. This is a serious concern as the access door/opening has to admit at least one astronaut in a bulky space suit with a tool kit.

The entry in the Table for complete engine removal may be the usual maintenance operation rather than attempting to remove individual components with connections to high pressure lines. The Aerojet design has reduced this operation to disconnect of four lines, separation of electrical connectors, and removal of the gimbal actuators. A list of separate operations for engine removal is given in Table 3.6.3-3. Another option is to design a propulsion module such as the OMS pod on the Space Shuttle for ready separation and removal from the vehicle. More extensive trade studies and development of maintenance timelines are needed to realistically assess any maintenance plan. In-space maintenance recommendations and conclusions are given in Table 3.6.3-4.

Table 3.6.3-2

OTV Engine Preliminary Design, Space Maintainable Components

7.5K lbf Thrust Engine

Component/Assembly	Comment
Hydrogen Main Shutoff Valve	Requires Access Near Engine/Vehicle Interface
Oxygen Main Shutoff Valve	Requires Access Near Engine/Vehicle Interface
Hydrogen Boost Pump (Low Press)	Requires Access Near Engine/Vehicle Interface
Oxygen Boost Pump (Low Press)	Requires Access Near Engine/Vehicle Interface
Hydrogen Tank Pressurization Valve	Requires Access Near Engine/Vehicle Interface
Oxygen Tank Pressurization Valve	Requires Access Near Engine/Vehicle Interface
Hydrogen Regenerator Bypass Valve	Design Dependent; May be a Bolt-On to a Manifold
Bypass Valve	Design Dependent; May be a Bolt-On to a Manifold
Gimbal Motors	Requires Access Near Engine/Vehicle Interface
Gimbal Actuators	Requires Access Near Engine/Vehicle Interface
Extendible Nozzle	Ready Access
Extendible Nozzle Deployment Motors	Requires Access Near Engine/Vehicle Interface
Extendible Nozzle Dep. Mechanism	Requires Access Near Engine/Vehicle Interface
Fuel Flowmeters	Design Dependent; May be Bolt-On to Low Press Boost Pump
Oxygen Flowmeters	Design Dependent; May be Bolt-On to Low Press Boost Pump
Controller	One or Two Removable Boxes with Cannon Plugs
Sensor Signal Conditioning Units	Designed to Allow Sensors to Remain In Place, Only Electronics Changed
Turbopumps	Design Dependent. Reliability Reqmts. may Require Welded Line
Igniters	Requires Access to Area Near Top of Thrust Chamber
Complete Engine	Requires Disconnect of 4 Propellant Lines, Electrical Cannon Plugs Disconnects, Gimbal Actuator Disconnects, Hinge Point Disassy Dependent on Access
Miscellaneous Hardware, Brackets, Wires, External Sensor Elements	

TABLE 3.6.3-3

OTV ENGINE REMOVAL OPERATIONS

- Engine Centered, Nozzle Extended
- Propellant Isolation Valves Closed
- Electrical Power Removed from the Engine
- Manual Operations:
    - Electrical Harness Disconnected, Cannon Plugs Capped, Harness Stowed
    - Extendible Nozzle Removed, Screw Assemblies Secured, Regen Cooled Nozzle Edge Protector Installed
    - Main Hydrogen Line Disconnected Below Shutoff Valve, Capped
    - Main Oxygen Line Disconnected Below Shutoff Valve, Capped
    - Hydrogen Tank Autogenous Pressurization Line Disconnected Below Shutoff Valve, Capped
    - Oxygen Tank Autogenous Pressurization Line Disconnected Below Shutoff Valve, Capped
    - Engine Handling Fixture Connected
    - Engine Out gimbal Actuator Disconnected
    - Control Gimbal Actuators Disconnected
    - Flex Lines Restrained, Upper Engine Covered with Protective Material
    - Prime Mover Connected to Engine Handling Fixture
- Engine Moved out of Engine Compartment

TABLE 3.6.3-4

OTV ENGINE IN-SPACE MAINTENANCE

Recommendations/Conclusions

- A realistic assessment of in-space maintenance costs is needed to determine the cost effectiveness of component changeout versus engine changeout

- An engine should not be designed for in-space maintenance if the capability is not going to be available during its operational life.

- Series production of an engine can be used to expand the maintenance capability as in-space facilities are expanded.

- The first version of an OTV may well be shuttle deployed and recovered with no in-space maintenance needed.

- The development of a propulsion module similar to the OMS Pod should be considered for the OTV as an alternative to engine and/or component changeout.

3.7 RELIABILITY AND RISK ASSESSMENT

All Aerojet engine designs must include a risk assessment per company policy. The usual form of this analysis is a YES-NO grid that identifies all technical and producibility risks plus a potential solution that can change all "No's" into "Yes." The program resources did not allow this formal analysis, and an abbreviated risk assessment is presented in Section 3.7.5.

Reliability was addressed with a preliminary FMEA and by identifying requirements and mapping an approach for a reliability program in the detail design phase.

3.7.1 Reliability Approach

The Aerojet approach to reliability task planning for the OTV engine is outlined in Table 3.7.2-1. It is based on both analytical and heuristic methods. This is a comprehensive approach as is appropriate to a very expensive manrated spacecraft.

The caveats and assumptions for a preliminary reliability analysis are given in Table 3.7.2-2. The engine reliability numbers are based on vehicle prime studies from Boeing and Martin-Marietta. It is assumed these numbers will be further scrutinized and refined as the vehicle development begins.

3.7.2 Failure Modes and Effects Analysis (FMEA)

An FMEA was completed under an earlier task in the OTV engine program. This FMEA was updated for the preliminary design and is included as Appendix A-1. This would be the start of an analysis that will be updated and expanded through detail design with the final version complete prior to production go-ahead.

3.7.3 Fail Operational/Fail Safe

Basic to the development of a reliability plan and to the engine design is the concept of "fail operational/fail safe". In essence, any component failure must produce either an operational though limited condition or a safe engine shutdown. This is implemented in the design by, for instance, selecting valves that will be spring loaded closed on power failure, i.e., main propellant shutoff valves; or fail in position, such as the turbine bypass valves. In the second example the engine can continue operation at the failure point but must be shutdown if an operational change is needed. Mechanical provisions are not sufficient for an engine of this complexity. They must be backed up by a health monitoring/control system that functions to assure that any

TABLE 3.7.2-1

OTV ENGINE RELIABILITY APPROACH

1. Statistical and Analytical
   - FMEA
   - Structural Analysis and Margins
   - Component Reliability Data Base
   - Reliability Number Assignment and Calculation

2. Heuristic
   - Fault Trees: No Catastrophes, No Cascades
   - Redundancies
   - Overdesign
   - Derating
   - Fabrication and Assembly Methods
   - Process Control

Both Must Assure:

- Fail Operational/Fail Safe

Table 3.7.2-2

OTV Engine Preliminary Design, Reliability Analysis

- NO RELIABILITY SUBTASK INCLUDED IN THE PRELIMINARY DESIGN TASK
- STUDY BASED ON EARLIER ATC WORK
- BASIC RELIABILITY REQUIREMENTS NOT CLEARLY DEFINED
- VEHICLE PRIME STUDIES PRESENT SEVERAL RELIABILITY NUMBERS

    MARTIN-MARIETTA: 0.9997 (BASED ON RL-10 RELIABILITY IN PRODUCTION)

    BOEING : 0.9975 SINGLE ENGINE
    : 0.99958 DUAL ENGINE

- ATC WORK COUPLED RELIABILITY WITH MANRATING
    - — ASSUMED OTV RELIABILITY EQUAL TO OR LOWER THAN ORBITER

failure is to continued operation or to shutdown. Table 3.7.3-1 lists the health management system functions that provide this capability.

3.7.4 Manrating

A manrated vehicle is assumed to have more stringent design requirements than an unmanned vehicle, but the definition of these requirements is difficult to find in a single source. Table 3.7.4-1 is an attempt to define manrating by the requirements found in various documents and programs.

Of concern to the OTV engine program are the need for redundant components and derating. Aerojet has assumed that redundancy is at the engine level as one engine must be capable of performing the mission. This has not been verified as a NASA position; and, perhaps, won't be until the development includes the first comprehensive reliability analysis. It is important to the design and needs to be formalized at an early date.

Derating is generally applied to the electronics but can be a concept used in the engine design as a more stringent design code can be called out and arbitrary margins increased to give the effect of derating.

3.7.5 Risk Assessment

Program risk is usually assessed against the technical, cost, and schedule goals of a program. For this preliminary design the cost and schedule risks are irrelevant although these risks would have to be considered in a development program. Technical risk is also divided among fabrication/producibility risk, performance risk, and operational risk. Each is discussed below. A major factor in the risk assessment is the Aerojet approach to this engine design. Aerojet has chosen a design approach that will offer the highest performance (specific impulse), best throttling performance (>10:1), and maximum operational flexibility (dual expander cycle, helium purge system). This is a higher risk approach than could have been selected, but the high payoff potential is considered in line with the program goals. In all cases there are lower risk fallback positions.

3.7.5.1 Fabrication/Producibility Risk

Producibility issues were presented in Section 3.3.6. The focus there was on solving problems through process development and design. Risk assessment weighs the likely

TABLE 3.7.3-1

FUNCTIONS PROVIDED BY HEALTH MONITORING/CONTROL SYSTEM

- Pre-Flight Engine Check-Out
- Engine Start Sequencing
- Closed-Loop Engine Control
- Red-Line Limit Detection
- Red-Line Limit Response
- Pink-Line Limit Detection
- Pink-Line Limit Response
- Trend Data Collection and Storage
- Engine Stop Sequencing
- Life Prediction and Maintenance Scheduling
- Communication with Vehicle Controller

Table 3.7.4-1

OTV Engine Preliminary Design, What is Man Rating?

Defined As:

- **Redundancy Plus High Reliability**
- **Systems which Tolerate Two Failures**
- **Reference: JSCM 8080 "Manned Spacecraft Criteria & Standards", Standard Number 12A (Rev 5/19/75)**
  - **Exempts Structure, TPS & Pressure Vessels**
  - **For Critical Flight Vehicle Subsystems, Minimum Redundancy Requirement is "Fail Safe"**
- **Failure of Any Single Subsystem without Loss of Life or Vehicle**
- **Shuttle Design Philosophy (for some Components)**
  - **Fail Ops/Fail Ops/Fail Safe**
- **Shuttle Payload Requirements**
  - **Fail Safe/Fail Safe**

success of such activities. A formal risk assessment will be completed during detail design complete with YES-NO grid and a realistic evaluation of the likely success. At this time the criteria was to present a summary of the risks with proposed solutions. This is done in Table 3.7.5-1 for the thrust chamber. One result of this assessment was to address some of the concerns in the FY-88 program as Task C.4, Baffled Injector Fabrication. It is intended that the items marked with an asterisk in the table will be verified under this task order.

A good portion of the fabrication/producibility risk will have to be reconsidered as the design progresses. With the design halted at the preliminary level there is not enough information for a high confidence analysis. To this point, however, none of these risks appears to be insoluble. There are design solutions available for all identified risks.

3.7.5.2 Performance Risk

Engine predicted performance was given in Section 3.1.5. There is high confidence that a specific impulse of >481 seconds is possible at MR=6 and a chamber pressure of 2000 psi. The NASA goal of 490 seconds specific impulse is not attainable within the envelope (area ratio limitation) and without breakthroughs in materials that will allow a chamber pressure of 3000 psia or higher. Improvements in combustor operation are of no help as the energy release efficiency using the I-triplet element and the baseline chamber configuration is very nearly 100%. A change in nominal operating point from MR=6 to an optimum performance point near MR=4 would provide some additional performance but would require larger hydrogen tanks which adversely effects the vehicle design. A full 10:1 throttling ratio is also considered low to moderate risk. Individual component performance requirements are either state-of-the-art or, as is the case with the turbopumps, require improved capability that is being developed under NASA contract.

3.7.5.3 Operating Risk

The ability of the design to meet operational requirements is a function of the reliability of the engine, the duty cycle, and health management system effectiveness. Reliability is discussed earlier in this section. The duty cycle or representative duty cycles are not well defined although they will likely include;

- LEO to GEO and return to LEO
- LEO to moon transfer orbit and return to LEO
- Moon transfer orbit to moon landing, moon takeoff and transfer to LEO

Table 3.7.5-1
Risk Assessment

MAJOR RISK AREA	TYPE OF RISK	SOLUTION
1. INJECTOR DESIGN	NEW, HIGHLY COMPLEX DESIGN	COMPLETE THE DESIGN AND FABRICATE
2. BAFFLE PLATE COOLING	INJECTOR-WALL COMPATIBILITY	1) DESIGN VERIFICATION TESTING* 2) DETUNE ELEMENT*
3. REGEN CHAMBER CHANNEL MACHINING	REQUIRES SEVERAL THOUSAND MACHINING STEPS WITHOUT ERROR (10 MIL CHANNELS)	VERIFY MACHINING CAPABILITY ON COPPER BILLETS*
4. REGEN CHAMBER NiCo CLOSEOUT	ELECTRO FORMING OF AN ALLOY. OPERATION COMPLETED SUCCESSFULLY ONLY ONE TIME	VERIFY EF-NiCo CLOSEOUT ON FULL SIZE CHAMBER
5. BONDING OF PLATELET FACEPLATE TO INJECTOR BODY	NICKEL TO INCONEL DIFFUSION BOND (BAFFLE PLATE ASSEMBLY INSTALLED AFTER FACE PLATE BONDED	1) DEMONSTRATE BOND STRENGTH* 2) CHANGE INJECTOR BODY TO NICKEL
6. BAFFLE PLATE PRESSURE CAPABILITY	ZrCu DIFFUSION BOND WITHOUT EXCE-EDING TEMPERATURE LIMITS ON BAFFLE PLATES	1) DEMONSTRATE BOND STRENGTH AT COMPATIBLE TEMPERATURE* 2) BUILD AND PRESSURE TEST A ZrCu BAFFLE PLATE
7. FACE WELD OF BAFFLE ASSEMBLY	GEOMETRY RESTRICTIONS ON E.B. PATH TO FACE PLATE	BRAZE BAFFLE ASSEMBLY IN PLACE

* IN FY88 PROGRAM

Table 3.7.5-1

Risk Assessment (Cont.)

COMPONENT/PROCESS	TYPE RISK	SOLUTION
8. OXYGEN COOLED NOZZLE	1) HIGH TEMP OXYGEN COMPATIBILITY	1) MATERIALS SELECTION (MONEL OR NASA-Z COPPER)
	2) LIGHTWEIGHT DESIGN	2) TUBE BUNDLE, MATERIALS SELECTION (BERYLLIUM?)
9. RADIATION COOLED NOZZLE EXTENSION	1) LOADS NOT DEFINED TO ESTABLISH DESIGN	1) DELAY UNTIL LOADS ARE AVAILABLE OR OVERDESIGN
	2) GAS SEAL LEAKS	2) USE SOLUTION(S) DEVELOPED BY SOLID ROCKET COMPANIES (ASPC)
10. NOZZLE RETRACTION MECHANISM	FAILURE TO DEPLOY OR RETRACT, OR STOP IN INTERMEDIATE POSITION	SELECTION OF SIMPLE (JACKSCREW) REDUNDANT (3 MOTOR) DRIVE-SYSTEM WITH LIMIT SWITCHES
11. TURBINE BYPASS VALVES	DEVELOPEMENT OF SERVO VALVES FOR HOT H_2, WARM O_2 SERVICE	ADAPT VALVES USED BY SOLID PROPELLANT SYSTEM FOR OTV-E USE

The minimum number of engine starts/shutdowns for any of these missions is four, with five or six more likely. Engine operating time will range from about a minute (orbit adjust maneuvers) to 25 minutes (moon transfer orbit). Throttled operation should be minimal except for the moon landing and takeoff. If this requires five minutes of throttled operation in both landing and takeoff, then the less than full thrust operation time will very likely be less than 10% of total engine operation time. Such a proportion would be very desirable as engine life is substantially reduced by operating below 500 psia chamber pressure for prolonged periods. Once the duty cycle is defined the actual life prediction can be made, but at this time the number of start/stop cycles and full thrust operating time are expected to be compatible with life goals of 500 starts and/or 20 hours of operation. A summary of engine life factors is given in Table 3.7.5-2.

The engine development has to solve the problems inherent in a tank head start, bootstrap from pumped idle, and throttling operation. These are summarized in Table 3.7.5-3. They were discussed more fully in Section 3.1. All are considered solvable and design solutions are incorporated in this preliminary design. The effectiveness of the design solutions will have to be proven by a combination of analysis and testing after the design is completed.

TABLE 3.7.5-2

ENGINE LIFE FACTORS

Factor	Concern
1. Number of Starts	1. Service life is an inverse function of number of starts. Quantification requires a knowledge of the engine thermal conditions over a nominal start cycle. This is not known at this time.
2. Operating Time Below 500 psia Pc	2. Copper thrust chamber parts revert to annealed properties at the high temperatures expected at low chamber pressure operation. This should be a time limited operation.
3. Overthrust Operation	3. Overthrust is a capability and a potential mission safety factor but subjects copper chamber parts to high temperatures that cause reversion to annealed properties. It should be a time limited operation.
4. Exposure to High Radiation Levels	4. After several years operation in space above and through the Van Allen belts the total radiation exposure could damage some engine parts.

Table 3.7.5-3
Risk Assessment

ENGINE OPERATION MODE	TYPE RISK	SOLUTION
TANK HEAD START	1) OFF MIXTURE RATIO OPERATION	1) HYDROGEN IDLE VALVE (SERVO)
	2) FLAME OUTS, PRESSURE SPIKES ON RELIGHT	2) DELAY IGNITION UNTIL PRESSURES STABILIZE
	3) INABILITY TO BOOTSTRAP TO PUMPED MODE	3) CHANGE SYSTEM TO AVOID A TANK HEAD START
PUMPED IDLE	NO CAPABILITY TO BOOTSRAP	TPA, FLOW CIRCUIT DESIGN CHANGES
THROTTLE OPERATION	1) EXCESSIVE BULK TEMP. RISE AT LOW Pc	1) REGENERATOR IN BYPASS MODE
	2) OPERATION THROUGH THE OXYGEN 2-PHASE REGION	2) OXYGEN GASIFIER, BACKPRESSURE VALVES
	3) CONTROL CAPABILTY, RESPONSE	3) PROPER FEEDBACK, RATES, CONTROLLER DESIGN, VALVE DESIGN
	4) SYSTEM TIME-LAGS, DISJUNCTION AT START	4) OXYGEN GASIFIER SHOULD REDUCE OXYGEN CIRCUIT LAG
	5) OPERATIONAL LIFE AT LOW Pc	5) MAY BE AN ENGINE LIMIT
OVERTHRUST	1) REDUCED CONTROL RESPONSE	1) VARY THRUST TO HOLD MR CONSTANT
	2) REDUCED SAFETY FACTORS	2) TIME LIMITED OPERATION

Table 3.7.5-3 Risk Assessment (Cont.)

COMPONENT/PROCESS	TYPE RISK	SOLUTION
8. OXYGEN COOLED NOZZLE	1) HIGH TEMP OXYGEN COMPATIBILITY	1) MATERIALS SELECTION (MONEL OR NAS-Z COPPER)
	2) LIGHTWEIGHT DESIGN	2) TUBE BUNDLE, MATERIALS SELECTION (BERYLLIUM?)
9. RADIATION COOLED NOZZLE EXTENSION	1) LOADS NOT DEFINED TO ESTABLISH DESIGN	1) DELAY UNTIL LOADS ARE AVAILABLE OR OVERDESIGN
	2) GAS SEAL LEAKS	2) USE SOLUTION(S) DEVELOPED BY SOLID ROCKET COMPANIES (ASPC)
10. NOZZLE RETRACTION MECHANISM	FAILURE TO DEPLOY OR RETRACT, OR STOP IN INTERMEDIATE POSITION	SELECTION OF SIMPLE (JACKSCREW) REDUNDANT (3 MOTOR) DRIVE-SYSTEM WITH LIMIT SWITCHES
11. TURBINE BYPASS VALVES	DEVELOPEMENT OF SERVO VALVES FOR HOT H_2, WARM O_2 SERVICE	ADAPT VALVES USED BY SOLID PROPELLANT SYSTEM FOR OTV-E USE

4.0 SUMMARY OF RESULTS

A preliminary design is successful only if the separate components are found to be packagable into a functional form capable of meeting the original design requirements as an integrated unit. The component integration task demonstrated that packaging was possible despite the many valves, the separate turbopumps, and the use of two heat exchangers. The functional capability of the design was verified in the power balance work. This design can meet operating requirements using the basic components proposed. By this standard it is a successful design ready for further development in a detail design program.

For any device as complex as a pump fed rocket engine, a preliminary design will reveal limitations and also offer opportunities for innovative design solutions. The comments below offer some limitations with possible design solutions and design solutions that neatly avoided apparent limitations:

- The "best effort" approach to completing the work packages produced in all cases an output commensurate to the task scope. This is recommended as an excellent approach for task order programs of limited resources.

- The power balance code was changed considerably over the program. This was one subtask where added resources would have been useful for additional component optimization and for mapping of more of the operating envelope. The start of detail design should emphasize in-depth work on, and with the power balance.

- Thermal limitations were expected above 2000 psia chamber pressure (Pc), but the life limiting temperatures below 500 psia Pc were unexpected. The regenerator effectiveness was limited by the throat gas side wall temperature. A design change that could alleviate this would be to use a ceramic coating covering two to three inches of the critical throat area. NASA LeRC is investigating such coatings as this report is being written, so the technology should be in hand for the Pathfinder program.

- More flexibility in the hydrogen circuit power balance is possible if a higher hydrogen temperature out of the baffles is attained. A proposed high temperature baffle plate design is one solution. This would add 300°F to the hydrogen gas that could either increase the control margin or raise the chamber pressure.

- The full 10:1 throttling is possible with the design at MR = 6. Operation below 500 psia chamber pressure is recommended at MR = 5 and with time limits to extend chamber life.

- Tank head start is possible with some reservations on repeatability and start time. Confidence in its practicality will require actual demonstration testing.

- Bootstrapping from tank head idle to the normal operational range is possible due to the use of the low pressure boost pumps. They are a vital part of the design.

- The Aerojet performance prediction is for a delivered specific impulse in the range of 481.7 to 484 lbf-sec/lbm for the baseline design. This prediction uses a somewhat conservative methodology.

- A design alternative for a very long life engine is to design rated thrust at the thermal minimum point. The resulting engine would be slightly larger and heavier but would have significantly higher operating margins. it would also be capable of operating at a thrust 60% greater than nominal.

- Aerojet's investigation of thrust chamber materials found that the SCM company's GLIDCOP AL-15 is potentially superior to NASA-Z in terms of life cycle fatigue. The results were presented to NASA-LeRC personnel, but more work is needed to validate the life cycle fatigue calculation methods and the material itself.

- A major benefit of the dual expander cycle with separate turbopumps is the elimination of a helium purge system. This greatly simplifies system plumbing and saves weight in excess of the actual engine weight.

- The dependence of the engines on electrical power is such that a backup battery system is recommended so that engine operation can continue despite a power interruption between the full cells on the OTV and the engines.

- Engine openings through the aerobrake will be approximately 5 feet by 12 feet. This means that opening covers (doors or sliding panels) will be large, heavy, and will require a complicated actuation system on the order of that for the landing gear of a jet aircraft.

- The primary and engine-out gimbal concept used to attain the ±20° specified has several advantages: lighter weight, simplified extendible nozzle design, and smaller engine envelope. It's disadvantage is to require an additional actuator.

- The engine-out gimbal design led to a concept for engine changeout that is simple and compatible with operations in space.

- The nominal engine weight of 298.1 lbm (gimballed components) reflects the multitude of engine valves and heat exchangers needed for the throttling and tank head start operations. These requirements drive the weight up from a possible 210 lbm to the 298 lbm calculated.

- All major producibility concerns are being addressed in the on-going OTV engine development program.

- Valve selection was straightforward, but there are two areas where additional work is warranted: 1) valve weight reduction, and 2) turbine bypass valve development.

- Trade studies favored an all electrical valve set except for the back pressure valves which are line pressure operated. This eliminated the need for pneumatic or hydraulic systems.

- A benefit of the LOX/GH_2 heat exchanger (HEX) is the increase in density of the cooler hydrogen entering the injector. This allows for smaller injector passages and should save some weight. The effect on the element performance needs to be evaluated in the detail design phase.

- The design problem for the oxygen phase change within the HEX was not solved. There is a good possibility that a special channel geometry could counteract the film boiling and assure a predictable phase change without any liquid phase flow downstream from the HEX.

- The oxygen cooled nozzle is complex enough to warrant a separate development program.

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APPENDIX A-1

OTV ENGINE DESIGN

FAILURE MODES AND EFFECTS ANALYSIS (FMEA)

D011.8 A-COVER PGS

OTV DUAL PROPELLANT EXPANDER CYCLE ENGINE
FAILURE MODES AND EFFECTS ANALYSIS

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
Propellant Inlet Shutoff Valves (LO_2/LH_2) (Series redundant/one valve on each side of interface)	1. 1 unit fails to close.	1. No effect - series redundant.		
	2. Both units fail to close.	2. No effect - close tank isolation valve.		X
	3. 1 or 2 unit(s) fail to open.	3. No engine operation W/O EVA and manual operation of valve.	X	
	4. 1 unit fails partially open or closed.	4. Engine may operate at partial thrust but can still be throttled, started, shutdown.		X
	5. Response time out of tolerance	5. No effect on closing (series redundant). No effect on opening due to to (relatively) long start time operation (tank head idle mode).		
Oxidizer Turbopump Assembly	1. Loss of LO_2 to hydrostatic bearings (contamination).	1. Bearings have redundant lubrication passages - partial loss may result in bearing wear. Significant or total loss may cause ignition or melting and HM directed shutdown. (Pump surfaces can tolerate rubbing friction for TBD msec without catastrophic failure).	X	X
	2. Cavitation or bubble/gas ingestion due to blocked or restricted inlet (contamination, or partial inlet valve failure, or failure in propellant acquisition system.)	2. Reduced pump life, reduced thrust. HM directed shutdown due to pump overspeed. Restart is possible if overspeed did not reach design limit.	X	X

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
	3. Turbine blade cracks, failure, or damage.	3. HM directed shutdown or reduced thrust operation.	X	X
Fuel Turbopump Assembly	1. Loss of LH_2 to hydrostatic bearings (contamination).	1. Bearings have redundant lubrication passages-partial loss may result in bearing wear, reduced pump life. Total loss will result in pump failure.		X
	2. Cavitation or bubble/ingestion due to blocked or restricted inlet (contamination or partial inlet valve failure or failure in propellant acquisition system).	2. Reduced pump life, reduced thrust. HM directed shutdown due to pump overspeed. Restart is possible if overspeed did not reach design limit.	X	X
	3. Turbine blade cracks, failure, or damage.	3. HM directed shutdown or reduced thrust operations.	X	X
Oxidizer Boost Pump	1. Pump fails to operate.	1. Requires tank head start - questionable reliability	Uncertain	
Concept A: Individual boost pumps for each OTV engine.	2. Pump operates at reduced flow or pressure.	2. Engine MR, thrust, Isp are effected.		X
Concept B: 2 ea. boost pumps feeding engine cluster inlet manifold.	3. 1 pump to operate or operates at reduced flow or pressure.	3. No effect - parallel redundant.		
	4. Both pumps fail to operate.	4. If successfully started on tank head pressure autogenous system will give normal operation.	Uncertain	

OTV DUAL PROPELLANT EXPANDER CYCLE ENGINE
FAILURE MODES AND EFFECTS ANALYSIS

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
High Pressure Pump Discharge Line(s)	1. Leaks or cracks at lines, seals, or joints.	1. Engine operates off MR. Lowr thrust and Isp		X
Hydrogen system only	2. Catastrophic structural failure or rupture.	2. Engine fails to opeate due to insufficient fuel.	X	
Oxygen system only	3. Catastrophic structural failure.	3. Possible fire, low order explosion.	X	
Igniter System	1. 1 unit fails to operate.	1. No effect - fully redundant system.		
(Dual-redundant spark gap type elements. Inductive ignition coils, and electronics - may utilize hydrogen cooled ceramic element in lieu of igniter valves.)	2. Both units fail to operate.	2. No engine ignition.	X	
	3. Spark delay on startup.	3. Pressure spike and HM directed engine shutdown.	X	
Oxygen Turbine Bypass Valve	1. Valves fail to open, close, or partially operate; or response time out of limits.	1. Loss of throttling capability on affected engine. In multiple engine concept, other engines can gimbal and/or throttle to correct or null thrust vector. Health monitoring system can detect valve operational failure and adjust fuel throttling valve(s) to prevent catastrophic MR and resulting TCA damage.		X
	2. Valve surfaces have rubbing contact or fretting.	2. Possible ignition depending on magnitude and duration of friction. HM directed engine shut-down.	X	TBD

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
	3. Valve has internal leakage.	3. Magnitude of leakage may affect throttling capability and MR.		X
Hyrogen Turbine **Bypass Valve**	1. Valve fails to open, close, or partially operate; or response time out of limits.	1. Loss of throttling capability on affected engine. In multiple engine concept, other engines can gimbal and/or throttle to correct or null thrust vector. Health monitoring system can detect valve operational failure and adjust oxidizer bypass valve to prevent catastrophic MR and resulting TCA damage.		X
	2. Valve has internal and/or external leakage.	2. Magnitude of leakage may affect throttling capability and MR, reduced Isp.		X
	3. Failure of valve actuating coil control.	3. No effect. Redundant operating coils/ controls.		
Thrust Chamber Assembly	1. Leaks or cracks on TCA inner walls.	1. System may operate off MR, reduced thrust and Isp.		X
	2. Leaks or cracks on TCA outer walls.	2. System operates off MR, reduced thrust and Isp. Shutdown if leak detection system is installed.	X	X
	3. Major rupture or substantial foreign object damage.	3. Engine shutdown by HM system.	X	
	4. Plugged cooling channels (contamination).	4. May reduce engine life, otherwise no effect on performance.		X

OTV DUAL PROPELLANT EXPANDER CYCLE ENGINE
FAILURE MODES AND EFFECTS ANALYSIS

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
Baffle Plate	1. Burn through in outer wall.	1. May affect MR, Isp depending on magnitude.		X
	2. Plugged cooling channels (contamination).	2. May reduce baffle life - otherwise no effect on performance.		X
Injector	1. Plugged orifices (contamination).	1. May affect MR, Isp, thrust. Worst case (loss of fuel film cooling) may result in reduced engine life (thermal limits exceeded) or could cause TCA burn through. HM directed shutdown.	TBD	X
Engine Controller/ Health Monitoring System	1. HM system erroneously predicts engine/system, component degradation or failure.	1. Engine operates at lower thrust, Isp, or off MR. May be directed to shutdown.	X	X
Concept A: Supervisory or limited control.	2. Controller fails to adjust engine operation based on actual (non-catastrophic fault).	2. Reduced engine life. May cause HM directed shutdown.	X	X
Concept B: Full authority control (redundant multiple channel).	3. Partial failure of controller, or loss of instrumentation parameter.	3. No effect - controller is redundant channel, can synthesize missing parameter or substitute nominal performance value.		
	4. Total failure of controller or instrumentation group.	4. Loss of engine operation. Propellant shut-off valves close.	X	

OTV DUAL PROPELLANT EXPANDER CYCLE ENGINE
FAILURE MODES AND EFFECTS ANALYSIS

COMPONENT OR SUBSYSTEM	FAILURE MODE	SYSTEM EFFECT	SYSTEM FAILURE FULL	SYSTEM FAILURE PARTIAL
Instrumentation Transducer	1. Transducer fails totally, calibration shifts, or transducer gives incorrect readings and/or transients.	1. No effect - health monitoring system can synthesize parameter or substitute nominal performance value. Most transducers will be redundant dual element design; health monitoring system can "vote out" incorrect reading by rate-of-change transient computations (transducer drop outs or erratic transients) and by comparison with other combustion transducer parameters.		No effect
Gimbal System	1. Gimbal actuators fail to extend or retract, insufficient stroke, erratic operation or slew rate.	1. Unaffected engines (multiple engine scenario) can compensate by selective gimbaling or throttling to null or correct thrust vector.		X
Radiation Cooled Nozzle	1. Burn throughs, coating cracks, structural damage from handling, foreign objects/meteors.	1. Reduced thrust, Isp - otherwise no effect on engine system operation (throttling and/or gimbal system can compensate at system (multiple engine) level in worst case. Possible local heating of aerobrake.		X
Nozzle Retraction	1. Fails to retract.	1. May prevent aerobraking. Jettison if system has that capability.	TBD	
	2. Fails in intermediate position.	2. May prevent aerobraking. Jettison if system has that capability.	TBD	
Nozzle Extension	1. Fails to extend.	1. Engine can be operated. No gimbal capability. Some local heating of aerobrake.		X

APPENDIX A-2

HEALTH MONITORING SENSORS

TABLE A-2.1
HEALTH MONITOR AND CONTROL SENSOR LIST

Component	Sensor	Location	Function			Wt.	PWR.
			Control	HM-1	HM-2		
Hydrogen Turbopump Assy.	Z1 Shaft Axial Displ.	Inside First Stage Turbo-pump Assy.		X			
	Z2 Shaft Radial Displ.			X		2.5 for 3 Functions	10 VDC 30MA for 3 Functions
	S1 Speed			X			
	Z3 Shaft Axial Displ.	Inside First Stage Turbo-pump Assy. 90° from Z1		X			
	Z4 Shaft Radial Displ.			X		2.5 for 3 Functions	10 VDC 30MA for 3 Functions
	S2 Speed				X		
	P-1 Pump Discharge Pres.	To Line @ Pump Disch.	XX			0.5 Oz.	10 VDC 15MA
	T-1 Pump Discharge Temp.	To Line @ Pump Disch.	XX			0.7 Oz.	10 VDC 15MA
	F-1 Pump Flow		XX			12 Oz.	28 VDC 20MA
	Z5 TPA Vibration	On T-P Housing Between Brgs.		X		.07 Oz.	15 VDC 4MA
Hydrogen Turbopump Assy.	Z6 Shaft Axial Displ.	Inside 2nd Stage Turbo-pump Assy.		X			
	Z7 Shaft Radial Displ.			X		2.5 Oz. for 3 Functions	10 VDC 30MA for 3 Functions
	S3 Speed			X			
	Z8 Shaft Axial Displ.	Inside 2nd Stage Turbo-pump Assy. 90° from Z3		X			
	Z9 Shaft Radial Displ.			X		2.5 Oz. for 3 Functions	10 VDC 30MA for 3 Functions
	S4 Speed				X		
	P2 Pump Interstage Pres.	To TPA Between Stages			X	0.5 Oz.	10 VDC 15MA
	T2 Pump Interstage Temp.	On TPA Between Stages			X	0.7 Oz.	10 VDC 15MA
	P3 Turbine 2nd Stage Inlet Pres.	On TP Housing Between Stages			X	0.5 Oz.	10 VDC 15MA
	T3 Turbine 2nd Stage Inlet Temp.				X	0.3 Oz.	10 VDC 15MA

TABLE A-2.1
HEALTH MONITOR AND CONTROL SENSOR LIST (CONT.)

Component	Sensor	Location	Function			Wt.	PWR.
			Control	HM-1	HM-2		
Hydrogen Turbopump Assy.	P4 Pump Inlet Pres.	In Line @ Pump Inlet			X	0.5 Oz.	10 VDC 15MA
	P5 Turbine Inlet Pres.	In Line @ Turbine Inlet		X			
	P6 Turbine Disch Pres.	In Line @ Turbine Outlet			X		
	T4 Turbine Inlet Temp.	In Line @ Turbine Inlet	XX			0.7 Oz.	10 VDC 15MA
	T5 Turbine Disch. Temp	In Line @ Turbine Disch.			X	0.7 Oz.	10 VDC 15MA
	F2 Turbine Inlet Flow	In Line @ Turbine Inlet			X	12 Oz.	28 VDC 20MA
	T6 Pump Inlet Temp.	In Line @ Pump Inlet			X	0.7 Oz.	10 VDC 15MA
Hydrogen Boost Turbo-pump. Assy.	Z10 Bearing Outer Race Deflectometer	On Bearing Outer Race .005 In. Nominal Clearances		X		0.8 Oz.	10V 20MA for 2 Sensors
	Z11 Bearing Outer Race Deflectometer			X		0.8 Oz.	2 Sensors
	Z12 Boost Pump Accel.	On Boost Pump Housing Be-tween Brgs.			X	0.07 Oz.	15 VDC 4MA
	P7 Boost Pump Inlet Pres.	In Line @ Pump Inlet			X	0.5 Oz.	10 VDC 15MA
	T-7 Boost Pump Inlet Temp.	In Line @ Pump Inlet			X	0.7 Oz.	10 VDC 15MA
	F3 Boost Turbine Inlet Flow	In Line @ Turbine Inlet			X	12 Oz.	28 VDC 20MA
Oxygen Turbopump Assy.	Z13 Shaft Axial Displ.	Inside Turbo-pump Assy.		X			
	Z14 Shaft Radial Displ.			X		2.5 Oz. for 3 Functions	10 VDC 30MA for 3 Functions
	S5 Speed			X			
	Z15 Shaft Axial Displ.	Inside Turbo-pump Assy. 90° from Z5		X			
	Z16 Shaft Radial Displ.			X		2.5 Oz. for 3 Functions	10 VDC 30MA for 3 Functions
	S6 Speed				X		

TABLE A-2.1
HEALTH MONITOR AND CONTROL SENSOR LIST (CONT.)

Component	Sensor	Location	Function			Wt.	PWR.
			Control	HM-1	HM-2		
Oxygen Turbopump Assy.	P8 Pump Disch. Pres.	In Line @ Pump Disch.	XX			0.5 Oz.	10 VDC 15MA
	T7 Pump Disch. Temp.	In Line @ Pump Disch.	XX			0.7 Oz.	10 VDC 15MA
	F4 Pump Flow		XX			12 Oz.	28 VDC 20MA
	Z17	On T.P. Housing Between Bearings		X		0.07 Oz.	15 VDC 4MA
Oxygen Turbopump Assy.	P9 Pump Inlet Pres.	In Line @ Pump Inlet			X	0.5 Oz.	10 VDC 15MA
	P10 Turbine Inlet Pres.	In Line @ Turbine Inlet		X			
	T8 Turbine Inlet Temp.	In Line @ Turbine Inlet	XX			0.7 Oz.	10 VDC 15MA
	T9 Turbine Disch. Temp.	In Line @ Turbine Disch.			X	0.7 Oz.	
	T10 Pump Inlet Temp.	In Line @ Pump Inlet			X	0.7 Oz.	
	F5 Turbine Inlet Flow	In Line @ Turbine Inlet			X	12 Oz.	28 VDC 20MA
Oxygen Boost Turbo-pump Assy.	Z18 Bearing Outer Race Deflectometer	On Bearing Outer Race .005 In. Nominal Clearance		X		0.8 Oz.	28 VDC 20MA for 2 Sensors
	Z19 Bearing Outer Race Deflectometer			X			
	Z20 Boost Pump Accel.	On Boost Pump Hsg. Between Brgs.			X	.07 Oz.	15 VDC 4MA
	P11 Boost Pump Inlet Pres.	In Line @ Pump Inlet Thermal Isolation Req'd by Tube Conn.			X	0.5 Oz.	10 VDC 15MA
	T11 Boost Pump Inlet Temp.	In Line @ Pump Inlet			X	.7 Oz.	10 VDC 15MA
	F6 Boost Turbine Inlet Flow	In Line @ Turbine Inlet			X	.12 Oz.	28 VDC 20MA
Combustion Chamber	P12 Chamber Pressure	Combustion Chamber	XX			0.5 Oz.	10 VDC 15MA

TABLE A-2.1

HEALTH MONITOR AND CONTROL SENSOR LIST (CONT.)

Component	Sensor	Location	Function			Wt.	PWR.
			Control	HM-1	HM-2		
Injector	P13 Fuel Inlet Pressure	At Injector Fuel Inlet		X		0.3 Oz.	10 VDC 15MA
↓	T12 Fuel Inlet Temp.	↓		X		0.7 Oz.	↓
	P14 Oxidizer Inlet Pressure	At Injector Oxidizer Inlet		X		0.5 Oz.	
	T13 Oxidizer Inlet Temp.	↓		X		0.7 Oz.	
Combustion Chamber	P15 Chamber Coolant Outlet Pres.	At Chamber Coolant Outlet		X		0.5 Oz.	
↓	T14 Chamber Coolant Outlet Temp.	↓		X		0.7 Oz.	
	P16 Chamber Baffle Coolant Outlet Pres.	At Transition Between Chamber Baffle Coolant			X	0.5 Oz.	
	T15 Chamber Baffle Coolant Outlet Temp.	↓			X	0.7 Oz.	
	T16 Chamber Throat Surface Temp.	Multiple Sensor Band Around Chamber Throat			X	TBD	TBD
Nozzle	P17 Fuel Coolant Inlet Pres.	At Nozzle Fuel Inlet Manifold		X		0.5 Oz.	10 VDC 15MA
↓	T17 Fuel Coolant Inlet Temp.	↓		X		0.7 Oz.	↓
	P18 Ox. Coolant Inlet Pres.	At Nozzle Ox. Inlet Manifold		X		0.5 Oz.	
	T18 Ox. Coolant Inlet Temp.	↓		X		0.7 Oz.	
	P19 Ox. Coolant Outlet Pres.	At Nozzle Ox. Outlet Manifold			X	0.5 Oz.	
	T19 Ox. Coolant Outlet Temp.	↓			X	0.7 Oz.	
Regen	T20 Fuel Regen Outlet Temp.	@ Fuel Regen Sec. Outlet Port.		X		0.7 Oz.	
↓	T21 Ox. Regen Fuel Inlet Temp.	Ox. Regen Inlet Port			X	0.7 Oz.	
	P20 Fuel Regen Inlet Pres.	Fuel Regen Inlet Port			X	0.5 Oz.	

TABLE A-2.1
HEALTH MONITOR AND CONTROL SENSOR LIST (CONT.)

Component	Sensor	Location	Function			Wt.	PWR.
			Control	HM-1	HM-2		
Fuel Turbine Bypass Valve	Z21 Position	On Valve Pintle Extension	XX			3.2 Oz. for 1 In. Stroke	28 VDC 20MA
Ox. Turbine Bypass Valve	Z22 Position		XX				
Hex. Bypass Valve	Z23 Position		XX				
Fuel Idle Valve	Z24 Position		XX				
Fuel Regen, Bypass Valve	Z25 Position		XX				
Nozzle Position	Z26 Position-Ret. Z27 Position-Ext.	Limit Switches at Limits of Travel	XX			0.3 Oz.	28 VDC 1MA
Engine Compartment	T22, T23, T24, T25 Temperature	In Engine Compartment 4 Places		X		0.02 Oz.	10 VDC 15MA
Engine Compartment	L1, L2, L3, L4 H2 Leak Detection	In Engine Compartment 4 Places		X		TBD	TBD
Engine Compartment	L5, L6, L7, L8 O_2 Leak Detection	In Engine Compartment 4 Places		X			
Total			16 + 16	46	31		

APPENDIX A-3

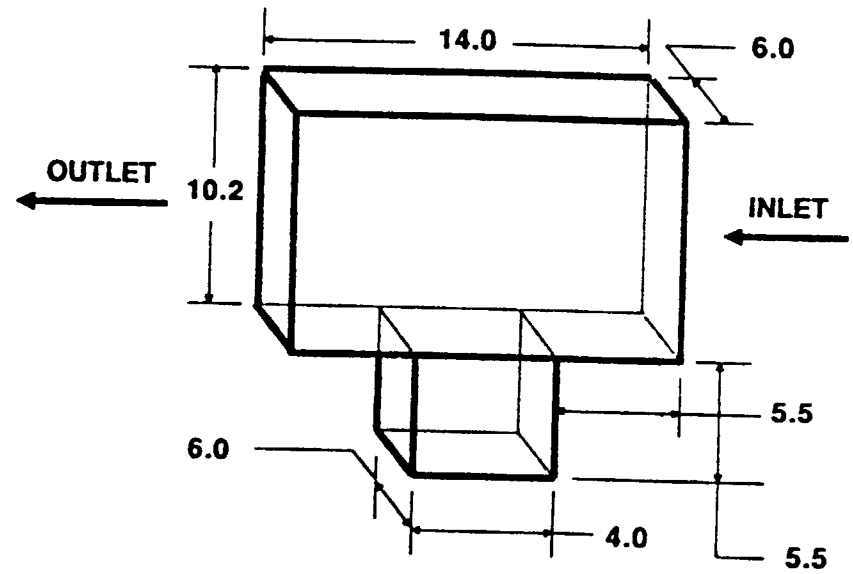
ENGINE VALVE SPECIFICATIONS AND ENVELOPE DRAWINGS

RPT/D0011.8-A-3

This appendix contains short specification sheets on each of the valves baselined for the OTV engine. The items specified should be sufficient for an initial vendor survey. A detailed specification would be developed during the detail design phase.

The envelope drawings are to be used by prospective vendors for rough sizing of the valves. They are also the envelopes used in the layout drawings for the engine. Some leeway would be expected to allow for a less-than-perfect match to the given envelope.

VALVE FUNCTION :	HYDROGEN MAIN SHUT-OFF
VALVE TYPE :	N.C.,ON/OFF,BALL;BFLY;BLADE
ACTUATION METHOD :	28 VDC MOTOR
EST. POWER REQ. :	165 WATTS
EST. WT. (+-20%):	8.5 LBS
RESPONSE TIME :	250 ms
FLUID MEDIA :	LH2
OP. PRESS.(psia):	0 TO 45
OP. TEMP. (F deg):	-400 to -422
DENS.(#/ft**3)(H/L):	4.34 / 4.34
FLOWRATE(#/s)(H/L):	2.63 / 0.26
PRESS. DROP(psi)(H/L) :	< 5
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	ALUMINUM ALLOY OR TITANIUM ALLOY
COMMENTS:	FAIL SAFE CLOSED ON LOSS OF POWER
LINE SIZE :	2.50 O.D.
LEAKAGE,INT(sccs of GHe):	10 @ 90 psia
CYCLE LIFE(Min):	500

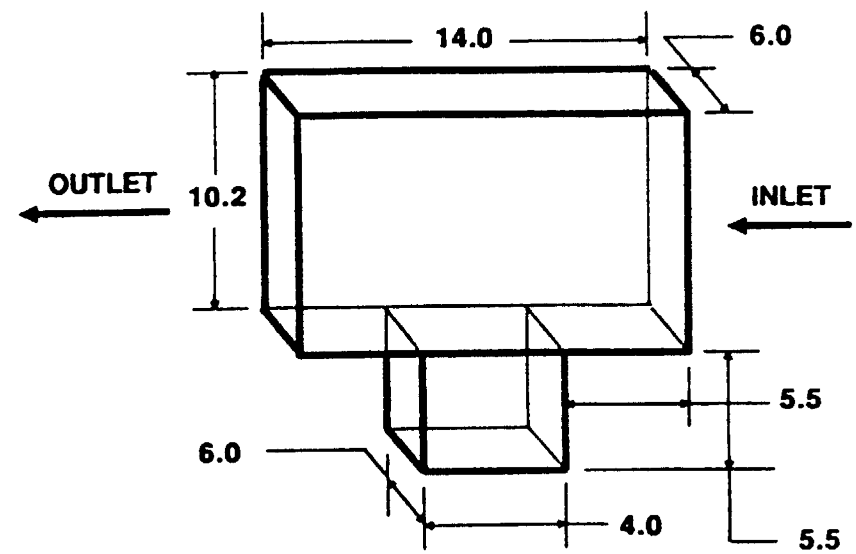


SHUT-OFF VALVE

VALVE FUNCTION :	OX MAIN SHUT-OFF
VALVE TYPE :	N.C.,0N/OFF,BALL;BFLY;BLADE
ACTUATION METHOD :	28 VDC MOTOR
EST. POWER REQ. :	160 WATTS
EST. WT. (+-20%):	8.0 LBS
RESPONSE TIME :	250 ms
FLUID MEDIA :	LO2
OP. PRESS.(psia):	0 TO 45 PSIA
OP. TEMP. (F deg):	-298
DENS.(#/ft**3)(H/L):	71.2 / 71.2
FLOWRATE(#/s)(H/L):	16.24 / 1.45
PRESS. DROP(psi)(H/L) :	$\leq$ 5
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	ALUMINUM ALLOY

COMMENTS : FAIL SAFE CLOSED ON LOSS OF POWER

LINE SIZE :	2.25 O.D.
LEAKAGE,INT(sccs of GHe):	10 @ 90 psia
CYCLE LIFE(Min):	500

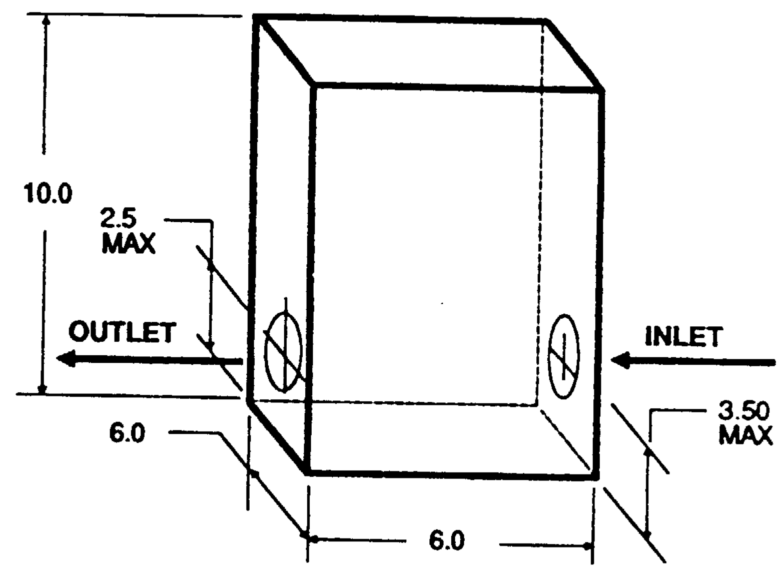


SHUT-OFF VALVE

VALVE FUNCTION :	OX TURBINE BYPASS
VALVE TYPE :	SERVO/PINTLE
ACTUATION METHOD :	28 VDC MOTOR
EST. POWER REQ. :	60 WATTS
EST. WT. (+-20%):	9.2 LBS
RESPONSE TIME :	100 ms
FLUID MEDIA :	WARM GO_2
OP. PRESS.(psia)(H/L):	4937 / 330
OP. TEMP. (F deg)(H/L):	400 /400
DENS.(#/ft**3)(H/L):	15.5 / 1.14
FLOWRATE(#/s)(H/L):	13.0 / 1.06
PRESS. DROP(psi)(H/L) :	2290 / 100
LEAKAGE ,EXT(sccs of GHe)	1×10^{-4}
MATERIALS OF CONSTRUCTION :	INCONEL 718 AND

OR MONEL 400 COMMENT : VALVE FAILS IN PLACE

LINE SIZE :	.750 O.D.
OPERATIONAL LIFE(Hrs):	20

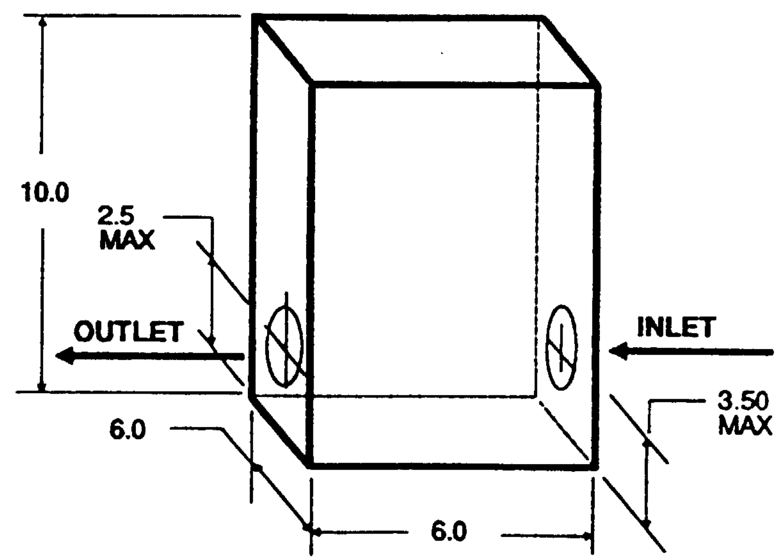


MODULATING VALVE

VALVE FUNCTION :	HYDROGEN TURBINE BYPASS
VALVE TYPE :	SERVO/PINTLE
ACTUATION METHOD :	28 VDC MOTOR
EST. POWER REQ. :	60 WATTS
EST. WT. (+-20%):	9.2 LBS
RESPONSE TIME :	100 ms
FLUID MEDIA :	HOT GH2
OP. PRESS.(psia)(H/L):	4889 / 298
OP. TEMP. (F deg)(H/L):	540 / 818
DENS.(#/ft**3)(H/L):	0.83 / 0.40
FLOWRATE(#/s)(H/L):	0.3 / 0.16
PRESS. DROP(psi)(H/L) :	2210 / 48
LEAKAGE EXT.(sccs of GHe):	1×10^{-4}

MATERIALS OF CONSTRUCTION : STAINLESS STEEL OR INCONEL 718 COMMENT : VALVE FAILS IN PLACE

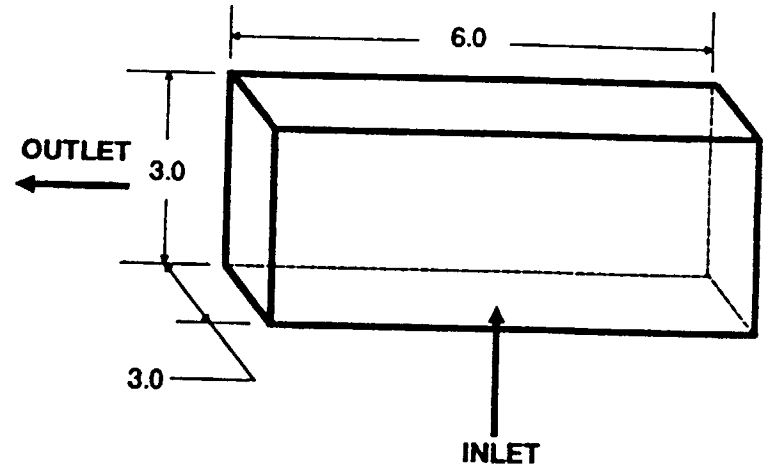
LINE SIZE :	.750 O.D.
OPERATIONAL LIFE(Hrs)	20



MODULATING VALVE

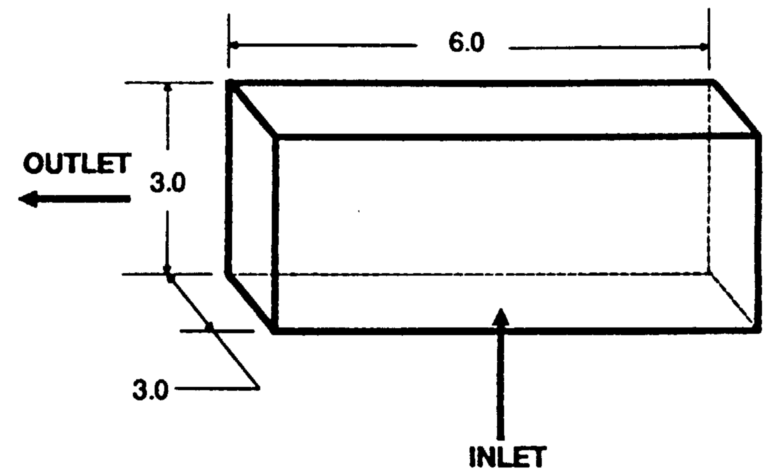
VALVE FUNCTION : OX TANK PRESSURIZATION
VALVE TYPE : N.C.,ON/OFF,POPPET
ACTUATION METHOD : 28 VDC SOLENOID
EST. POWER REQ. : 25 WATTS
EST. WT. (+-20%): 2.60 LBS
RESPONSE TIME : 200 ms
FLUID MEDIA : GO2
OP. PRESS.(psia): 0 to 3000
OP. TEMP. (F deg): 0 to 400
DENS.(#/ft**3)(H/L): 9.77 / .84
FLOWRATE(#/s)(H/L): 1.66 / .141
PRESS. DROP(psi)(H/L) : 2585 / 215
LEAKAGE , EXT(sccs of GHe): 1×10^{-4}
MATERIALS OF CONSTRUCTION : INCONEL 718 OR MONEL 400

LINE SIZE : .375 O.D.
CYCLE LIFE(Min): TBD



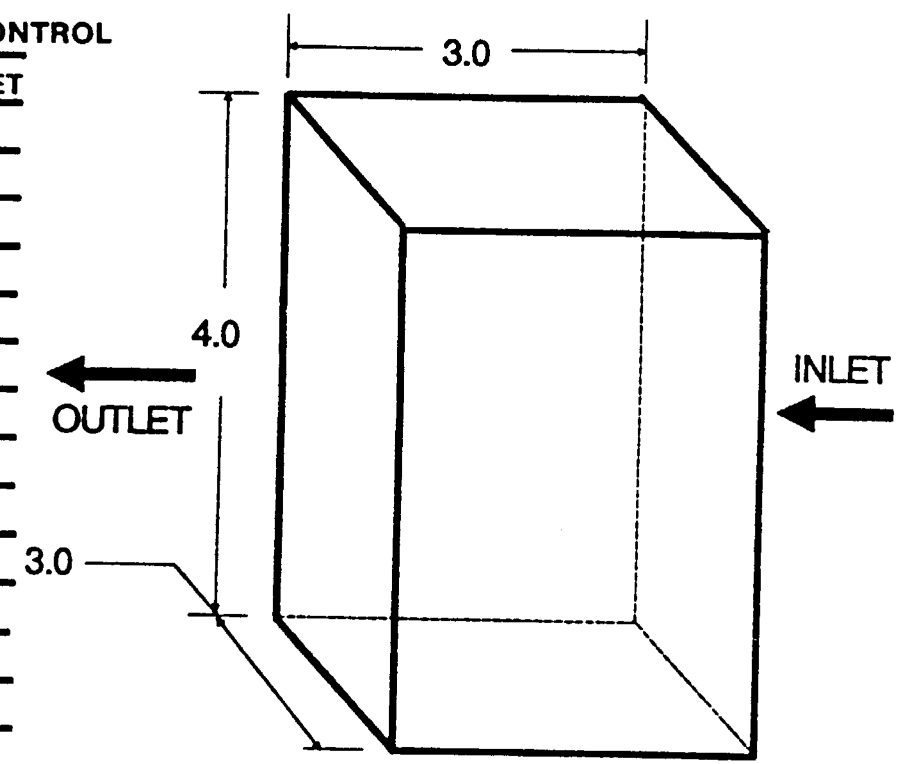
TANK PRESSURIZATION

VALVE FUNCTION :	HYDROGEN TANK PRESSURIZATION
VALVE TYPE :	N.C.,ON/OFF,POPPET
ACTUATION METHOD :	28 VDC SOLENOID
EST. POWER REQ. :	25 WATTS
EST. WT. (+-20%):	2.60 LBS
RESPONSE TIME :	200 ms
FLUID MEDIA :	GH2
OP. PRESS.(psia):	0 to 5000
OP. TEMP. (F deg):	0 to 800
DENS.(#/ft**3)(H/L):	0.49 / 0.04
FLOWRATE(#/s)(H/L):	.375 / .031
PRESS. DROP(psi)(H/L) :	2620 / 220
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	STAINLESS STEEL OR INCONEL 718
LINE SIZE :	.375 O.D.
CYCLE LIFE(Min):	TBD



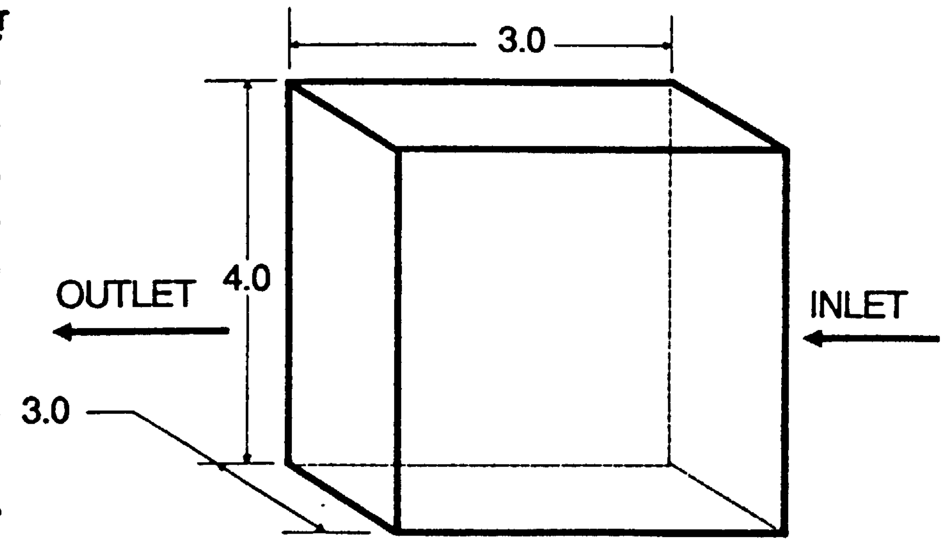
TANK PRESSURIZATION

VALVE FUNCTION :	HYDROGEN IGNITER CONTROL
VALVE TYPE :	N.C., OPEN/CLOSE,POPPET
ACTUATION METHOD :	28 VDC SOLENOID
EST. POWER REQ. :	24 WATTS
EST. WT. (+-20%):	1 LB
RESPONSE TIME :	20 ms
FLUID MEDIA :	GH2
OP. PRESS.(psia):	0-6000
OP. TEMP. (F deg):	-400 to 100
DENS.(#/ft**3)(H/L):	4.50/4.52
FLOWRATE(#/s)(H/L):	.040 / .0035
PRESS. DROP(psi)(H/L) :	TBS / TBS
LEAKAGE ,EXT(sccs of GHe):	1 X 10^{-4}
MATERIALS OF CONSTRUCTION :	S.S.
LINE SIZE :	.250 O.D.
CYCLE LIFE(Min):	500



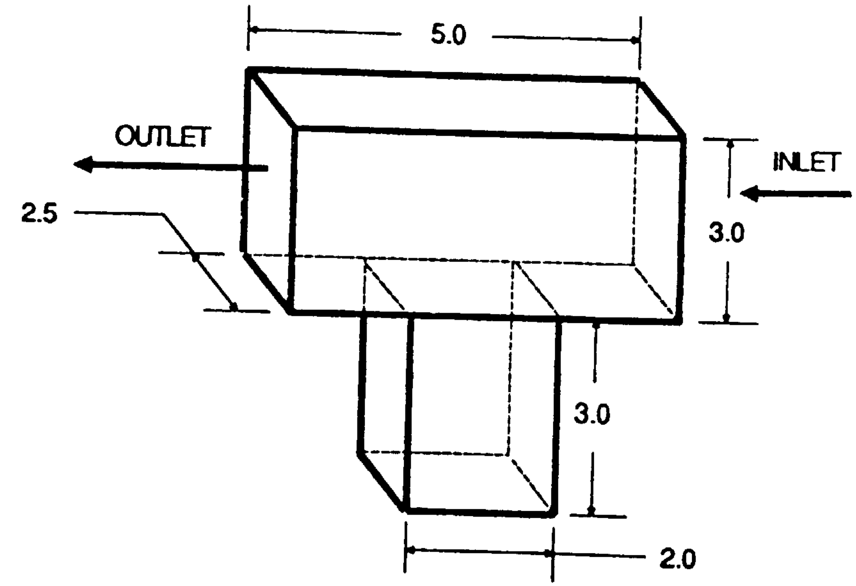
IGNITER VALVES

VALVE FUNCTION :	OX IGNITER CONTROL
VALVE TYPE :	N.C.,OPEN/CLOSED,POPPET
ACTUATION METHOD :	28 VDC SOLENOID
EST. POWER REQ. :	24 WATTS
EST. WT. (+-20%):	1 LB
RESPONSE TIME :	20 ms
FLUID MEDIA :	GO2
OP. PRESS.(psia):	0 -6000
OP. TEMP. (F deg):	-400 to 100
DENS.(#/ft**3)(H/L):	71.2/71.2
FLOWRATE(#/s)(H/L):	.145 / .0125
PRESS. DROP(psi)(H/L) :	TBS / TBS
LEAKAGE ,EXT(sccs of GHe):	1 X 10^{-4}
MATERIALS OF CONSTRUCTION :	MONEL 400 OR INCONEL 718
LINE SIZE :	.250 O.D.
CYCLE LIFE(Min):	500



IGNITER VALVE

VALVE FUNCTION :	HYDROGEN IDLE
VALVE TYPE :	PRESSURE BALANCE, OPEN/PARTIAL CLOSE
ACTUATION METHOD*:	LINE PRESSURE
EST. POWER REQ. :	N/A
EST. WT. (+-20%):	5.0 LBS
RESPONSE TIME :	250 ms
FLUID MEDIA :	GH2
OP. PRESS.(psia):	80 psid ACTUATE, 0 to 6000 psia
OP. TEMP. (F deg):	-360 to 540
DENS.(#/ft**3)(H/L):	.04
FLOWRATE(#/s)(H/L):	0.0 TO .26
PRESS. DROP(psi)(H/L) :	0.0 to 80
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	STAINLESS STEEL
LINE SIZE :	1.00 O.D.
OPERATIONAL LIFE(Hrs):	20



HYDROGEN IDLE VALVE

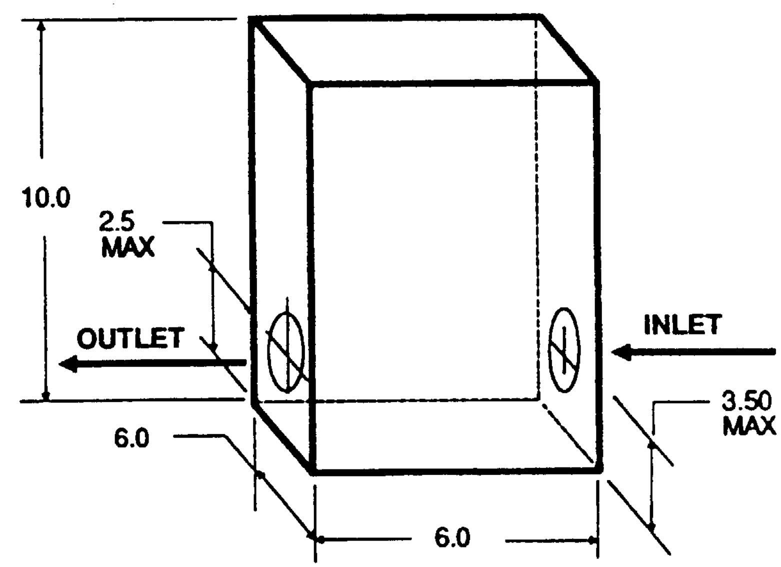
*THIS IS AN ALTERNATE VALVE SHOULD MIXTURE RATIO CONTROL ON STARTUP PROVE CRITICAL AND REQUIRE A MODULATING VALVE.

VALVE FUNCTION :	HYDROGEN REGEN, BYPASS
VALVE TYPE :	MODULATING POPPET
ACTUATION METHOD :	28 VDC MOTOR
EST. POWER REQ. :	60 WATTS
EST. WT. (+-20%):	9.2 LBS
RESPONSE TIME :	100 ms
FLUID MEDIA :	H2
OP. PRESS.(psia)(H/L)	5580 / 390
OP. TEMP. (F deg)(H/L)	-340 / -420
DENS.(#/ft**3)(H/L):	4.50 / 4.49
FLOWRATE(#/s)(H/L):	0.53 / 0.26
PRESS. DROP(psi)(H/L) :	75 / 5
LEAKAGE ,EXT(sccs of GHe)	1×10^{-4}
MATERIALS OF CONSTRUCTION :	ALUMINUM ALLOY

OR TITANIUM ALLOY

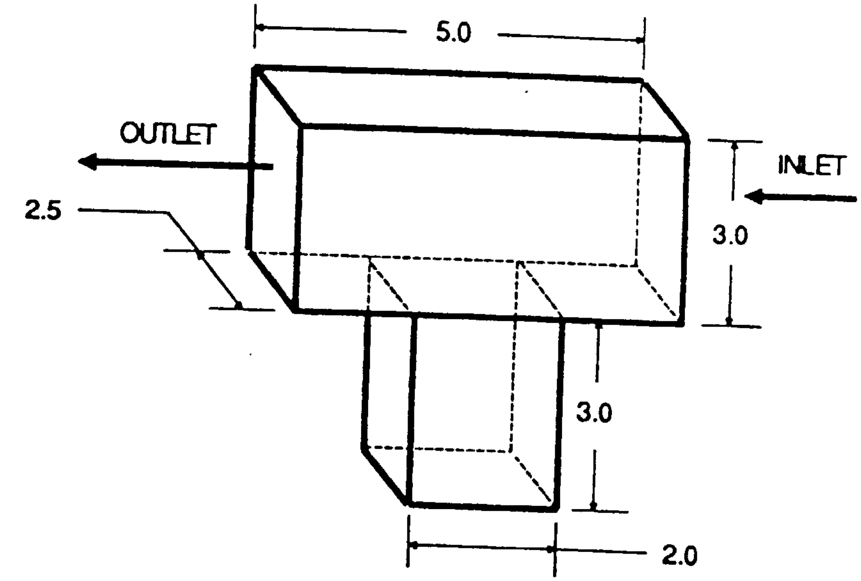
COMMENT : VALVE MUST FAIL OPEN

LINE SIZE :	1.00 O.D.
OPERATIONAL LIFE(Hrs):	20



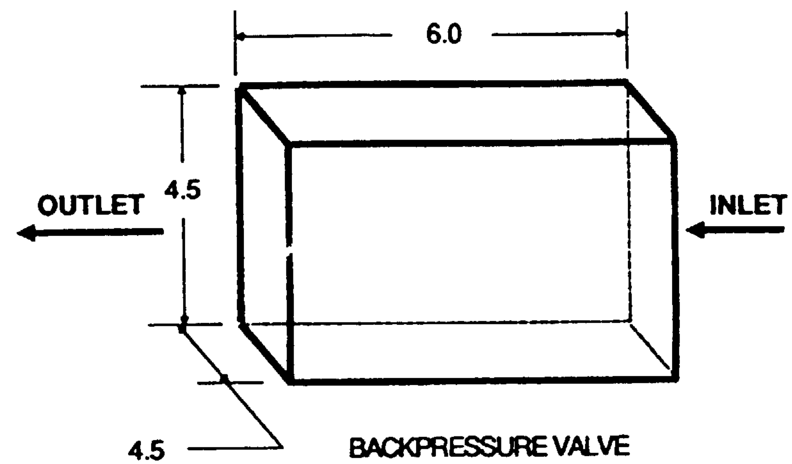
MODULATING VALVE

VALVE FUNCTION :	HYDROGEN IDLE
VALVE TYPE :	PRESSURE BALANCE, OPEN/PARTIAL CLOSE
ACTUATION METHOD*:	LINE PRESSURE
EST. POWER REQ. :	N/A
EST. WT. (+-20%):	5.0 LBS
RESPONSE TIME :	250 ms
FLUID MEDIA :	GH2
OP. PRESS.(psia):	80 psid ACTUATE, 0 to 6000 psia
OP. TEMP. (F deg):	-360 to 540
DENS.(#/ft**3)(H/L):	.04
FLOWRATE(#/s)(H/L):	0.0 TO .26
PRESS. DROP(psi)(H/L) :	0.0 to 80
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	STAINLESS STEEL
LINE SIZE :	1.00 O.D.
OPERATIONAL LIFE(Hrs):	20



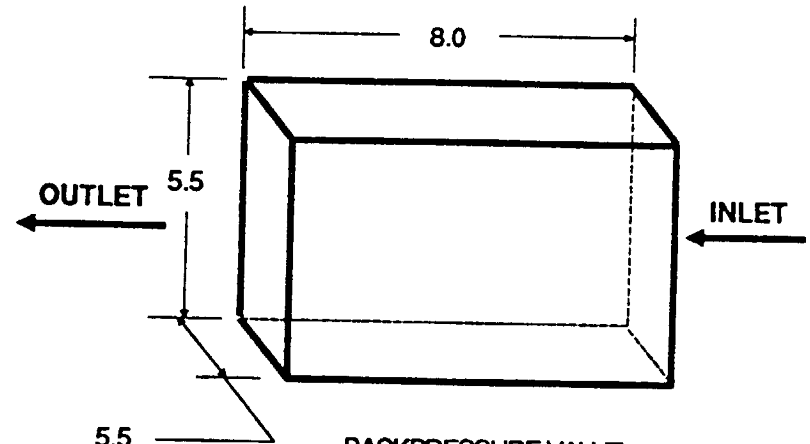
HYDROGEN IDLE VALVE

VALVE FUNCTION :	HYDROGEN BACK PRESSURE
VALVE TYPE :	N.C., OPEN/PARTIAL CLOSED ,POPPET
ACTUATION METHOD :	LINE PRESSURE
EST. POWER REQ. :	N/A
EST. WT. (+-20%):	3.0 LBS
RESPONSE TIME :	250 ms @ 500 psia
FLUID MEDIA :	GH2
OP. PRESS.(psia):	0 to 4889
OP. TEMP. (F deg):	50 to 818
DENS.(#/ft**3)(H/L):	0.83 / 0.04
FLOWRATE(#/s)(H/L):	2.63 / 0.26
PRESS. DROP(psi)(H/L) :	10.0 / 2.2
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	INCONEL 718 OR STAINLESS STEEL
LINE SIZE :	1.25 O.D.
CYCLE LIFE(Min):	500



BACKPRESSURE VALVE

VALVE FUNCTION :	OX BACKPRESSURE
VALVE TYPE :	N.C., OPEN/PARTIAL CLOSED ,POPPET
ACTUATION METHOD :	LINE PRESSURE
EST. POWER REQ. :	N/A
EST. WT. (+-20%):	5.0 LBS
RESPONSE TIME :	250 ms @ 500psia
FLUID MEDIA :	GO2
OP. PRESS.(psia):	0 to 4937
OP. TEMP. (F deg):	-260 to 400
DENS.(#/ft**3)(H/L):	15.5 / 1.14
FLOWRATE(#/s)(H/L):	16.24 / 1.45
PRESS. DROP(psi)(H/L) :	22.0 / 2.4
LEAKAGE ,EXT(sccs of GHe):	1×10^{-4}
MATERIALS OF CONSTRUCTION :	INCONEL 718 OR MONEL 400
LINE SIZE :	1.75 O.D.
CYCLE LIFE(Min):	500



BACKPRESSURE VALVE

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13. ABSTRACT *(Maximum 200 words)*
This document summarizes the preliminary design of the Aerojet version of the Orbit Transfer Vehicle main engine. The concept of a 7500 lbf thrust LO_2/GH_2 engine using the dual expander cycle for optimum efficiency is validated through power balance and thermal calculations. The engine is capable of 10:1 throttling from a nominal 2000 psia to a 200 psia chamber pressure. Reservations are detailed on the feasibility of a tank head start, but the design incorporates low speed turbopumps to mitigate the problem. The mechanically separate high speed turbopumps use hydrostatic bearings to meet engine life requirements, and operate at sub-critical speed for better throttling ability. All components were successfully packaged in the restricted envelope set by the clearances for the extendible/retractable nozzle. Gimbal design uses an innovative primary and engine out gimbal system to meet the ±20° gimbal requirement. The hydrogen regenerator and LOX/GH_2 heat exchanger uses the Aerojet platelet structures approach for a highly compact component design. The extendible/retractable nozzle assembly uses an electric motor driven jack-screw design and a one segment carbon-carbon or silicide coated columbium nozzle with an area ratio, when extended, of 1430:1. A reliability analysis and risk assessment concludes the report.

14. SUBJECT TERMS
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